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NIMBUS E and F

EXPERIMENT

INTERFACE REQUIREMENTS

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Section I

SUMMARY

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PURPOSE of this DOCUMENT

Achieve Early Design Compatibility

The achievement of early design compatibility and smooth, effective integration of the NIMBUS Spacecraft and experiments are primary requisites to program success. It is to these goals that this document is addressed.

Specific Purposes

The specific purposes of this document are:

- TO SPECIFY the criteria and requirements which allow for compatibility to be designed into the hardware initially.
- TO DEFINE what is done in system design and integration to ensure compatibility and minimize adverse effects upon the experiment and the spacecraft system.
- TO EMPHASIZE the need for the direct program participation of experimenters and experiment contractors, the extent that such participation is required, and why it is required.
- TO ESTABLISH the mechanism for interface control and the exchange of specific information.
- TO BEST UTILIZE experience gained from the design and integration of four NIMBUS Spacecraft.
- TO FAMILIARIZE new experimenters and experiment contractors with the NIMBUS Spacecraft and with the NIMBUS program.

Responsibility of Experimenter

It is the responsibility of the experimenter to see that this document is made available to and used by cognizant experiment design and management personnel.

Format and Organization of Handbook

The format and organization of this document have been devised to facilitate its use and stimulate user interest.

- THE TEXT is brief and to the point.
- GRAPHICS are used throughout.
- THE MATERIAL is covered in distinct sections.
- SEPARATORS identify subject areas for easy reference.
- THE NOTEBOOK BINDER APPROACH is used for easy addition and removal of pages.
- THE 8 1/2 X 11 PAGE is used to faciliate the use of copying machines.
- REVISED PAGES are identified by date of revision in upper right-hand corner of page.
- REVISIONS are identified by black line in outside border.

These features should be used to maximum advantage. The entire document should be read and it should receive frequent use. Specific sections can readily be copied for use by specialists.

NIMBUS PROGRAM

History

The NIMBUS program began in 1960 under the direction and management of the Goddard Space Flight Center (GSFC) of the National Aeronautics and Space Administration. Since the inception of the program, the spacecraft integration and test contractor has been the General Electric Company. Experiments and subsystems have been provided through the contributions of a variety of governmental agencies, industrial contractors, and universities. To date, the following seven spacecraft have resulted from the NIMBUS program:

- NIMBUS I and II (A and C)

  The first NIMBUS Spacecraft (NIMBUS A) was launched 28 August 1964 and designated NIMBUS I when it attained orbit. NIMBUS C was launched on 15 May 1966 and designated NIMBUS II upon attainment of orbit. At the time of preparation of this document, NIMBUS II was in its 22nd month in orbit and still operating successfully.

- NIMBUS B

  NIMBUS B was launched in May 1968. However, due to a booster failure, NIMBUS B never attained orbit.

- NIMBUS B_2 and D

  NIMBUS B_2 is scheduled for launch in early 1969 while NIMBUS D is projected into early 1970.

- NIMBUS E and F

  The sixth and seventh in the NIMBUS series of spacecraft are E and F. This document is applicable to experiments for these two spacecraft.

Objective

The primary objective of the NIMBUS program has been to develop a meteorological satellite system capable of meeting the research and development needs of the nation's atmospheric scientists and weather services. NIMBUS is now looked at to perform a broader applications mission. Overall objectives of the program are:

- TO PROVIDE AN EARTH-ORIENTED APPLICATIONS PLATFORM capable of supporting a wide variety of scientific sensors.
- TO VIEW THE ENTIRE EARTH on a daily basis for prolonged periods of time in keeping with research requirements.
- TO DEVELOP AND FLIGHT TEST advanced scientific sensors.
- TO COLLECT AND DISTRIBUTE DATA both for the study of atmospheric and earth processes and for immediate operational use, as required.

Launch

- WTR

  NIMBUS Spacecraft are launched from the Western Test Range (WTR), Vandenberg Air Force Base, California, at approximately midnight to achieve the desired orbit.

- VEHICLE

  The NIMBUS E and F launch vehicle is expected to be the Thorad/Agena D.

- ORBIT

  The present NIMBUS circular orbit altitude is approximately 600 nautical miles and the inclination is approximately 100 degrees. This results in a high noon sun-synchronous orbit. (The orbit precession rate is such that the sun remains approximately in the orbital plane.) Different orbit altitudes (from 300 to 1000 nm) are a possibility for future NIMBUS flights.

Ground Stations

- ULASKA AND ROSMAN

  The two Command and Data Acquisition (CDA) stations at Gilmore Creek, Alaska, and Rosman, North Carolina, utilize 85-foot diameter parabolic antennas to track, receive data from, and command the spacecraft. Ulaska acquires the spacecraft an average of 10 out of 14 orbits each day. Rosman acquires an average of two orbits a day (of the four missed at Ulaska) and two orbits for backup.

- NDHS AND NWSC

  Meteorological and engineering data transmitted by the spacecraft are received and partially processed at Ulaska and relayed over a wideband microwave link to the NIMBUS Data Handling System (MDHS) at GSFC and to the National Weather Satellite Center (NWSC) of the Environmental Science Services Administration at Greenbelt, Maryland. Rosman receives data from the spacecraft and transmits it in real time to the NDHS, where it is processed for use.

- NTCC

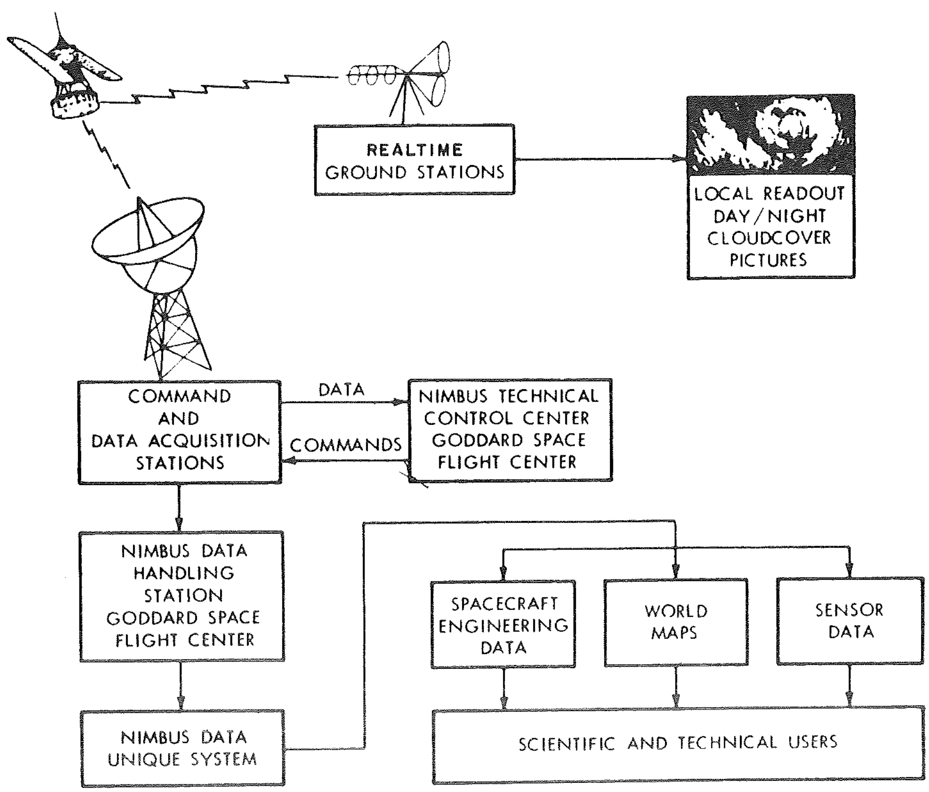
  The NIMBUS Technical Control Center (NTCC) at GSFC, responsible for command and control of the spacecraft, determines commands to be transmitted to the CDA stations over teletype or the wideband link.

- STADAN STATIONS

  In addition to ULASKA and ROSMAN,there are other STADAN (Satellite Tracking and Data Acquisition Network) stations which can be used for the recording of real time data.

- OTHER STATIONS

  Other stations (in addition to ULASKA and ROSMAN) may be used for tracking,data acquisition, and commanding the spacecraft to achieve more complete coverage if lower altitude orbits are flown.



NIMBUS PROGRAM DATA FLOW

SPACECRAFT CONFIGURATION

Basic NIMBUS Configuration

The NIMBUS Spacecraft (see top sketch, opposite) consists of a lower sensory ring structure and an upper housing separated by truss supports. The upper housing contains the Attitude Control Subsystem required for orientation of the spacecraft and provides solar paddle attachments. The rotating solar array is sun oriented. The sensory ring is composed of 18 bays for the location of modularized components of subsystems and experiments. An additional equivalent five bays can be provided in the cross beam area, if necessary. The attitude control system, sensory ring, and solar paddles are thermally independent. An active thermal control system (utilizing thermal shutters) is located on the outer periphery of the sensory ring.

Spacecraft Subsystems

- POWER

  The power available is -24.5 vdc regulated and -26.5 to -37.5 vdc unregulated. Regulated power is to be used by experiments. The average power available is approximately 225 watts (490 watts peak power).

- ATTITUDE CONTROL

  A 3-axis active control system provides earth orientation to a 1-degree pointing accuracy with low angular rates about each axis.

- TELEMETRY

  Versatile Information Processor (VIP) (analog and digital)

- COMMANDS

  High capacity (512 commands), capability for (up to 30) stored commands results in a versatile system.

- CLOCK

  Selected clock signals from 1 Hz to 1.6 MHz are available.

- DATA HANDLING

  - High data rate storage system (HDRSS)
  - Low data rate experiments (data handled by VIP) (VIP output recorded by HDRSS)

- VERY HIGH DATA RATE HANDLING

  A real time S-Band transmission system may be provided for transmitting data exceeding HDRSS and VIP data rate capabilities when the spacecraft is in view of a command and data acquisition (CDA) station. An additional possibility is the use of a very high data rate recording system in conjunction with the S-Band transmission system.

- THERMAL

  Combined active and passive, holding sensory ring temperature to 25 ± 10°C (and module temperatures to 25 ± 15°C). Heat dissipation capability of the different bays varies from 11 to 17 watts.

Location of Experiments

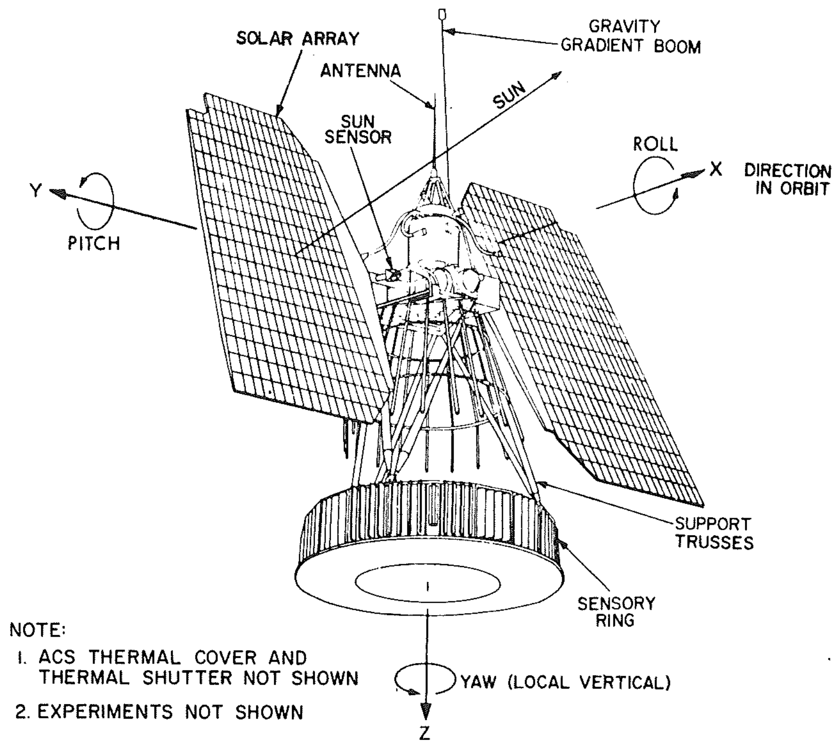
The NIMBUS sensory ring and the area inside the ring (the crossbeam area) are used for the location of earth viewing experiment optics and electronics. For these experiments, sensor optics could project as much as 24 inches below the ring, but in general, this is limited due to field-of-view and dynamic clearance considerations for other experiments and antennas. Experiments not requiring earth views may be mounted in other areas of the spacecraft such as on or inside the support trusses above the ring. Earth viewing experiments which cannot be mounted on or below the sensory ring could also be mounted in areas such as the support truss area, but may require deployment.

Sun Angle Variation

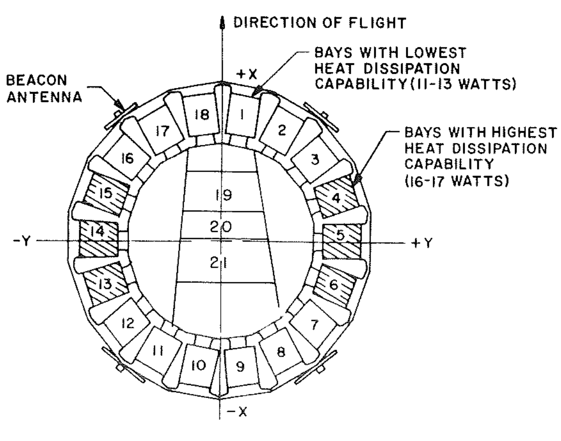
The angle between the orbit plane and the earth-sun line may be as large as 30 degrees. This factor must be considered in the design of experiments with special solar or space viewing requirements.

Bay Geometry and Module Size

The geometry and numbering of the bays of the NIMBUS sensory ring and the standard module size and nomenclature are shown below. These sizes must be adhered to for ring-mounted equipment and should be adhered to wherever possible for crossbeam mounted equipment.



NIMBUS CONFIGURATION



SENSORY RING CONFIGURATION

STANDARD NIMBUS MODULE SIZE AND NOMENCLATURE

(4/4 is called 4 over 4, etc)

4/4 (full bay)	6 x 8 x 13 inches
3/3	6 x 6 x 13 inches
2/2	6 x 4 x 13 inches
1/1*	6 x 2 x 13 inches
4/0 (or 0/4)	6 x 8 x 6.5 inches
3/0 (or 0/3)	6 x 6 x 6.5 inches
2/0 (or 0/2)	6 x 4 x 6.5 inches
1/0 (or 0/1)	6 x 2 x 6.5 inches

*Use of 1/1 Module Size to be avoided

ADVANCED INFORMATION

ADVANCED INFORMATION is required from experimenters IN ADDITION TO final design and test documentation. This information (outlined below) is required early in the program (see opposite page) to allow for orderly system definition, design, manufacture and test.

Experiment Information and Why it is Needed

STRUCTURAL/MECHANICAL
- Size, Shape, Weight and Center of Gravity
- Field-of-View (Earth and Space)
- Mechanical I/F Data (Drawings)
- Target Requirements
- Special Installation/Location Requirements
- Alignment Accessibility Requirements
- Purging Requirements

- Spacecraft Sensory Ring Configuration Equipment Location and Mounting
- Spacecraft and Adapter Mechanical/ Structural Design and Analysis
- Spacecraft Assembly Design
- Special Thermal Requirements
- Special Location Requirements

MODE OF OPERATION
- Duty Cycle
- Theory of Operation

- Orbital Sequence of Operations
  - Spacecraft Performance
  - Test Sequencing

POWER REQUIREMENTS
- Peak
- Average Orbital
- Standby
- High Voltage Requests (> 250 Volts)

- Power Profile
  - Thermal Design and Analysis
  - Power Management Design
  - Electrical Distribution

TELEMETRY REQUIREMENTS
- Number of Monitors
- Range of Measurement
- Required Accuracy/Sample Rate
- Analog or Digital
- T/M Calibrations

- Telemetry Directory
  - Telemetry System Design
  - Integration and Test Procedures
  - Operational Procedures

COMMAND REQUIREMENTS
- Number of Commands/Type
- Time of Occurrence/Sequences
- Stored Commands

- Command Directory
  - Command Distribution Design
  - Command Allocations (Integration and Test/Operations)

SENSOR OUTPUT DATA
- Analog or Digital
- Bit Rates
- Bandwidth or Frequency Response
- Voltage Levels/Source Impedances
- Signal Flow Diagrams
- Black Box Transfer Functions
- Signal Characteristics

- Data Evaluation
  - Software Requirements
  - Software Design and Development (Integration and Test/Operations)
  - Onboard and Ground Station Data Storage and Handling
  - Integration and Test/Operational Procedures

ELECTRICAL/ELECTRONIC
- Electrical I/F Data (Schematics)
- Clock Signal Requirements (Frequencies)
- Interconnect Requirements (Listing)
- Schematics and Special Wiring Rqmts
- Signal Characteristics & Impedances
- All Electrical Transient Characteristics

- S/C Electrical System Designs
  - Harness Design
  - Wire/Connector Selection
  - Shielding and Grounding Requirements
  - Clock Frequency Allocation and Distribution
  - Connector Type, Location, and Orientation

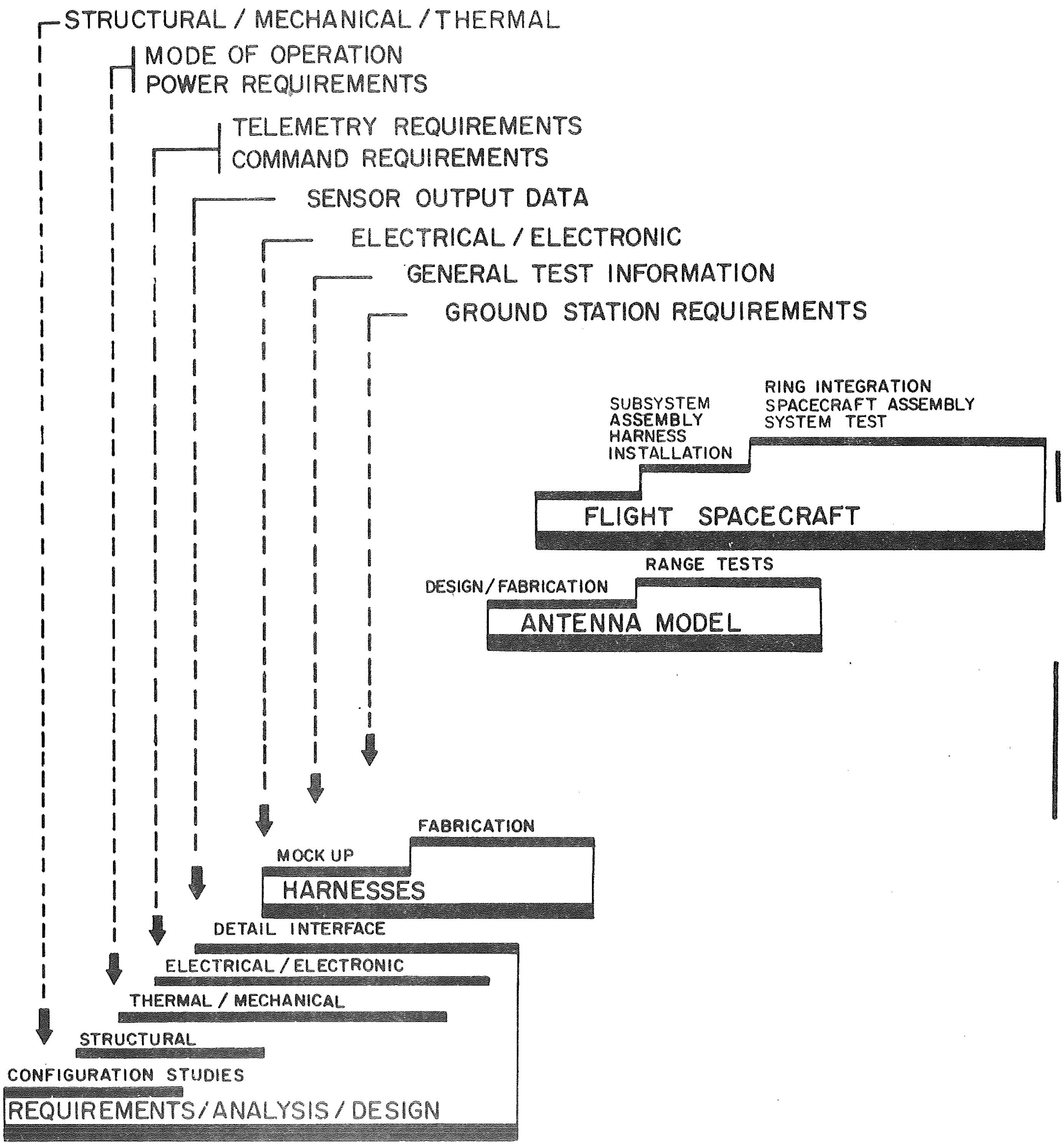
GENERAL TEST INFORMATION
- Bench Acceptance Test Procedure
- Acceptance Test Facility Requirements
- Calibration Curves
- Operating & Handling Instructions

- Test Facilities Readiness
  - Bench Acceptance/Integration Equipment Design and Setup
  - System Test Requirements and Procedures

GROUND STATION REQUIREMENTS
- Equipment and Facilities for Experiment Support (Test & Operational)

- Operational and Integration & Test Facility Ground Station Readiness and Checkout

ADVANCED INFORMATION RELATED TO TYPICAL PROGRAM EVENTS



ADVANCED INFORMATION (Cont'd)

Experiment Design Impact

Data provided by experimenters about their subsystem is used by many personnel in a myriad of ways in the integration of the experiment into the NIMBUS Spacecraft. These tasks are performed to verify compatibility of each experiment with the spacecraft and other experiments, which is a prerequisite for successful experiment operation. Since experimenter information is the major input to the integration process, this data must be as current and as accurate as possible.

NASA PROJECT MANAGEMENT

NASA project management and technical personnel use experiment design data throughout the program to perform the following tasks:

- NIMBUS Project and Spacecraft Management
- Provide Project Technical Direction
- Establish Design Criteria
- Design Review and Approval
- System Design
- Establish System, Subsystem & Test Requirements
- Assess Overall Spacecraft Design Compatibility
- Test Review and Approval

SPACECRAFT CONTRACTOR MANAGEMENT

Spacecraft contractor management and technical personnel put experiment design data to similar use. The following listings more specifically point up the technical importance of timely and accurate experiment design information:

SUBSYSTEM ENGINEERS

Subsystem engineers are the spacecraft contractor technical interface with experimenters, experiment contractors, and NASA GSFC Experiment Technical Officers. They use experimenter inputs in performing the following tasks:

- Experiment/Spacecraft Design Integration
- Developing Telemetry and Command Allocations
- Sensory Ring/Spacecraft Design
- Generating Test Requirements for Integration and System Tests
- Developing Software Requirements
- Evaluating Subsystem Performance (including the effects on and effects of other subsystems).

TEST AND EVALUATION PERSONNEL

Test and evaluation personnel use experimenter inputs to accomplish the following:

- Implement the Spacecraft Contractor Test Program with Test Procedures and Equipment
- Ensure Spacecraft Safety (through Design and Procedures)
- Verify the Status of the Spacecraft
- Develop System Evaluation Plans (including):
  - Evaluation Criteria
  - Evaluation Procedures
- Direct and Support System and Subsystem Evaluation (by assuring the required analysis is performed and documented through reports)
- Develop Software Requirements (to support test and evaluation)
- Evaluation of Test Results and Preparation of Test Reports

FLIGHT OPERATIONS PERSONNEL

Flight operations personnel utilize experimenter inputs to perform the following activities related to the in-orbit spacecraft:

- Develop Software Requirements for Inflight Evaluation
- Ensure Spacecraft Safety through Operational Procedures
- Optimize Payload Utilization
- Perform System and Subsystem Evaluations
- Issue Flight Evaluation Reports

Section II

MECHANICAL REQUIREMENTS

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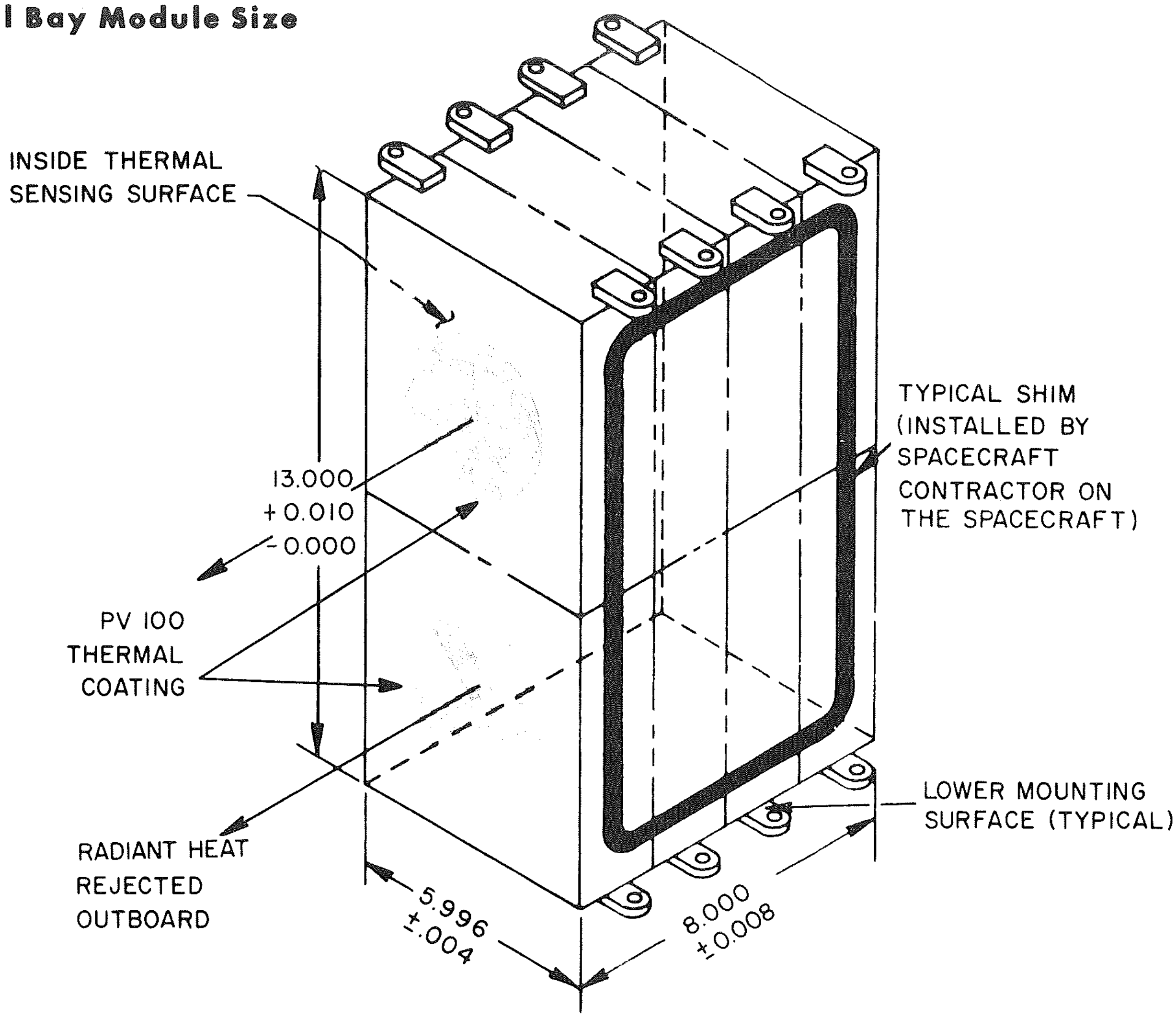
MECHANICAL INTERFACE

Full Bay Module Size



TYPICAL FULL-BAY NUMBUS MODULE (4/4)

Non-Modular Sized Components

The space available for non-modular sized components is very limited, consisting mainly of:

- EARTH-VIEWING AREA

  The area below the sensory ring is reserved, where possible, for sensor optics which must view the earth.

- CROSSBEAM AREA

  The area inside the sensory ring torus. A major portion of this area must be used for deep (long) components. In addition, there are difficulties in rejecting heat from this area.

Therefore, components must conform to the modular sizes to the maximum extent possible so that they can be located in one of the bays of the sensory ring.

MECHANICAL INTERFACE (Cont'd)

Mounting Tabs

Mounting tabs are required on bay-mounted modules to secure the module to the NIMBUS Sensory Ring in accordance with the Modular Interface Drawing (included as 2 sheets in this section).

- TABS MUST BE AT LEAST 1/4-INCH THICK
- USE ALL AVAILABLE TABS (16 for a 4/4 module) to minimize load concentrations
- ALL TABS MUST BE ATTACHED TO THE MODULES BY AT LEAST TWO SCREWS (No. 10-32)

  Inserts must be tapped into the module face to receive the screws. An insert such as a self-locking helicoil is required primarily for structural reasons and so that screws can be removed and reinserted many times. Both the insert and the screws must extend completely through the module face. Screws will have to be removed and reinserted for the following reasons:

  1. INSERTS will be used for attaching a handle to facilitate easy removal of modules during integration and test. In the past, modules could be damaged during removal from the bay since it was not possible to attach to the module, and thus the module had to be pried from the bay.
  2. TABS must be removable on full height modules to permit insertion of the module into the bay.

EXCEPTION TO REMOVABLE TAB RULE

Exception to the removable tab rule requires approval by the NASA spacecraft manager, and may be granted for hardware developed for previous NIMBUS Spacecraft. However, the following must be satisfied even if exception is granted:

a. All tabs must still be removable on either the upper or lower surface of all full height modules

b. At least two tapped holes into each tab utilizing inserts (No. 10-32) must be provided for easy removal of modules from the bays during integration and test. The tapped holes must extend completely through the tab.

- USE OF CONTINUOUS TABS as an alternate (see sketch opposite and the Modular Interface Drawing) requires early approval by the NASA GSFC spacecraft manager. If continuous tabs are used, they must be scalloped in accordance with the drawings to allow for spacecraft harness installation. At least four tapped holes per 1/0 module (two per tab) must be provided into the tabs. Tapped holes must utilize inserts (No. 10-32) as specified for individual tabs.

Materials

MATERIALS SPECIFICATION

Experimenters shall comply with section 407 of NASA publication S-450-P9 in the preparation and submittal of lists of materials used.

- MATERIALS LIST

  A list of all materials used in an experiment must be submitted to the spacecraft contractor for outgassing studies

- MODULE PACKAGE MATERIAL

  All experiment packages must be constructed of a lightweight material (such as magnesium or aluminum) that can be sheet metal, cast, machined, or a combination thereof. When sheet metal is used, care must be taken to provide adequate heat transfer to the spacecraft sink (the outboard surface of the sensory ring) and to provide sufficient strength for preloading.

- COMPATIBILITY WITH SPACECRAFT STRUCTURAL MATERIAL

  All material finishes must be compatible with the aluminum and magnesium spacecraft structure to inhibit electrolytic action

- ENVIRONMENTAL CONSIDERATIONS

  All material applications must be considered on the basis of expected environments, as well as on structural and electrical needs.

- ORIENTATION MARKING

  Any required orientation of a module must be clearly marked on that module and on the mechanical interface drawing. As a minimum, the outboard face of the bay-mounted module must be indicated.

Finishes

THE FOLLOWING FINISHES must be used for the indicated module materials:

- MAGNESIUM - Dow 23
- ALUMINUM - Alodine 600
- STEEL - Electroless Nickel

> NOTE
>
> No 1/1 Modules are to be used. Experiment modules requiring 1/4 of a bay should be packaged in a 2/0 (or 0/2) configuration as the location of a 1/1 module in a bay precludes the location of a battery (0/4 module) in that bay. NASA GSFC Spacecraft Manager approval is required for an exception to this rule.

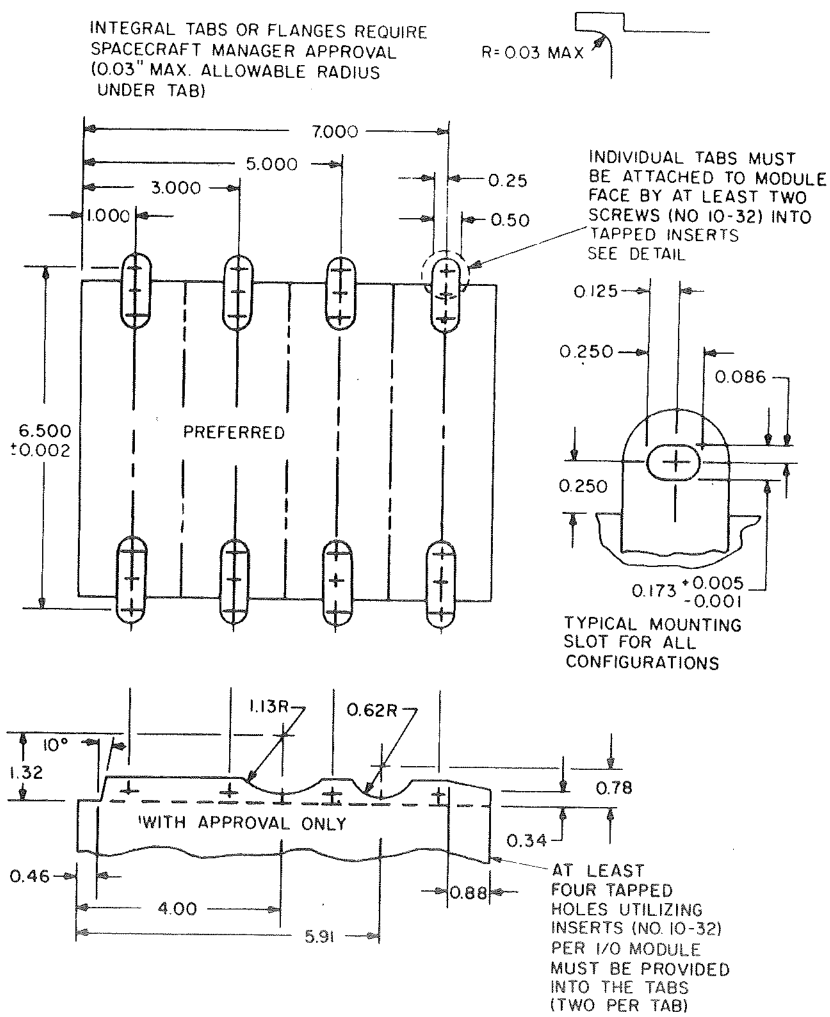
Note: Approval for use of steel must be obtained from GSFC Spacecraft Manager.

THE LOWER MOUNTING SURFACE OF MODULES (underside of tabs and flanges) must have a No. 32 finish.

(NOTE)

Unless otherwise specified, Dimensions are in inches. Tolerances not shown are:

2 Decimal ± 0.01
3 Decimal ± 0.005
On Angles $\pm 0.2^{\circ}$



MODULE INTERFACE CONFIGURATIONS

STRUCTURAL/INSTALLATION CRITERIA

Structural Restraints

- OPERATING RESTRAINTS - All experiments must be capable of operating in both a 1 G and 0 G field (any axis).

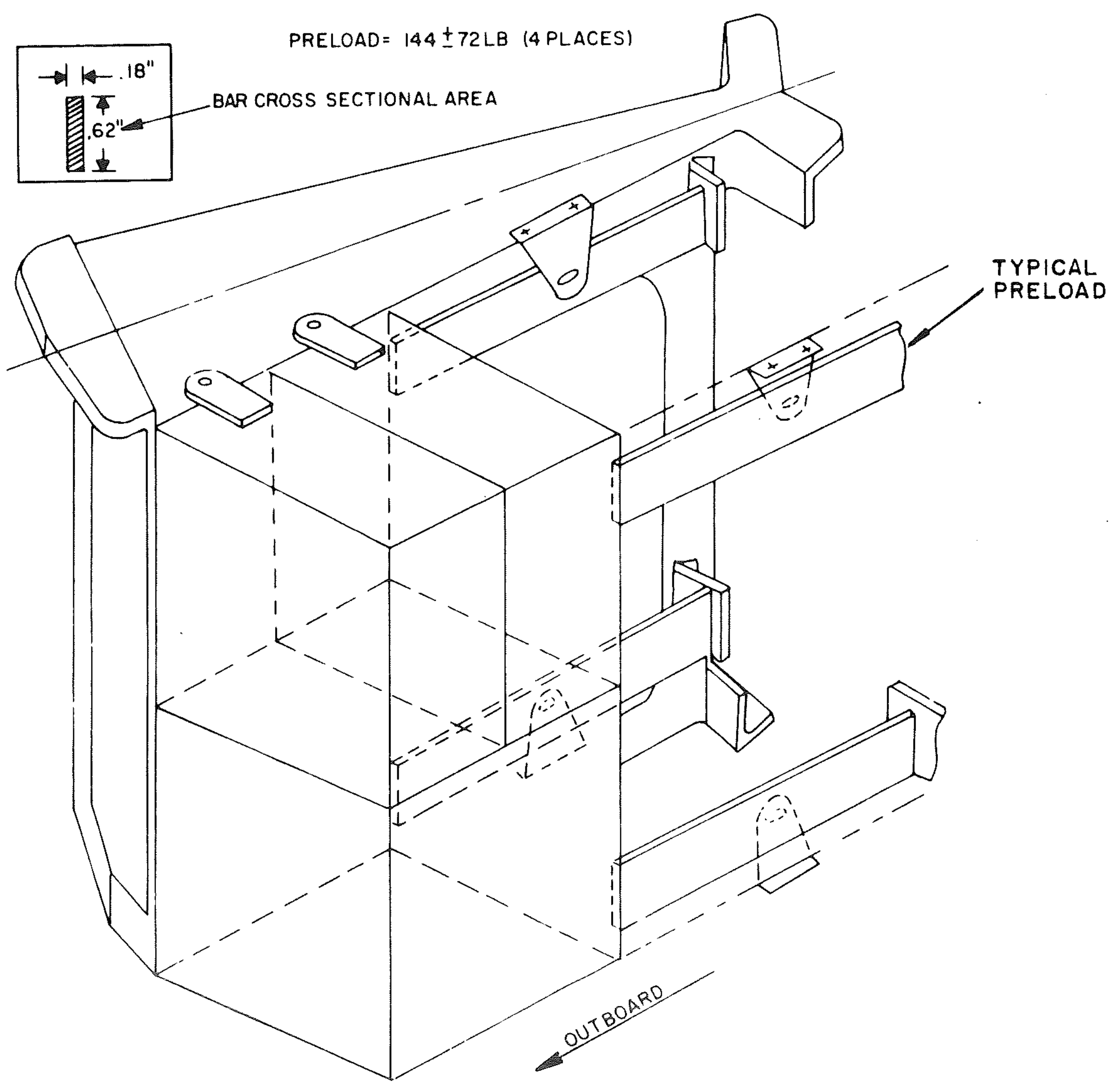
- MINIMUM NATURAL FREQUENCY - The minimum natural frequency of all externally mounted items and their supporting brackets shall be 100 cps to avoid coupling with the spacecraft structure.

- SENSOR AND EQUIPMENT INSTALLATIONS - Sensor and payload equipment installations must have capability for static and dynamic loading in all directions. Adequate load paths must be provided for shear and axial forces and bending and torsional moments. Yield and ultimate strengths must not be exceeded.

- NONREDUNDANT MOUNTING - Mounting arrangements must be nonredundant (statically determinant) to avoid overstressing or distorting sensors during installation.

- SENSOR ALIGNMENT - For sensors requiring alignment, the mounting arrangement must provide means of adjustment without compromising the shear carrying capability of the mounting. The final mounting point must be capable of sustaining loading in any direction of 30 G without shifting position.

- PRELOADING - Sensory ring modules are preloaded to provide good thermal contact (see sketch below). This preload is provided by four bars (0.62 x 0.18 inches each) with a capacity of 144 ±72 lb. Each module must be capable of sustaining this preload.



MODULE PRELOAD ARRANGEMENT

Installation Restraints (Accessibility)

- CLEANING LENSES - It must be possible to clean lenses without removal of the sensor or any portion of the sensor.
- PROTECTIVE COVERS - It must be possible to install and remove protective covers with one hand.
- STRONG LIGHT - Any sensors which are sensitive to strong light must be identified.
- PURGING OR CHARGING - For subsystems requiring purging or charging, the mechanical interface must be negotiated with the GSFC spacecraft manager and the spacecraft contractor.
- ACCESSIBILITY - Accessibility of items requiring alignment, test, and maintenance is mandatory.
- ATTACHMENT POINTS - Attachment points for AGE and test fixtures must be defined early and brought to the attention of the NASA GSFC spacecraft manager and the spacecraft contractor.
- ATTACHMENT BOLTS - It must be possible to withdraw attachment bolts without displacing the experiment sensor.
- CAPTIVE NUTS - If nuts are used, they must be captive.
- FASTENERS - Self-locking fasteners are required to prevent loosening during environmental testing and launch. The use of lock-tite is acceptable (except for types C & E).
- SENSOR INSULATION - If sensors require insulation, the experiment contractor may be required to supply mounting attachments.

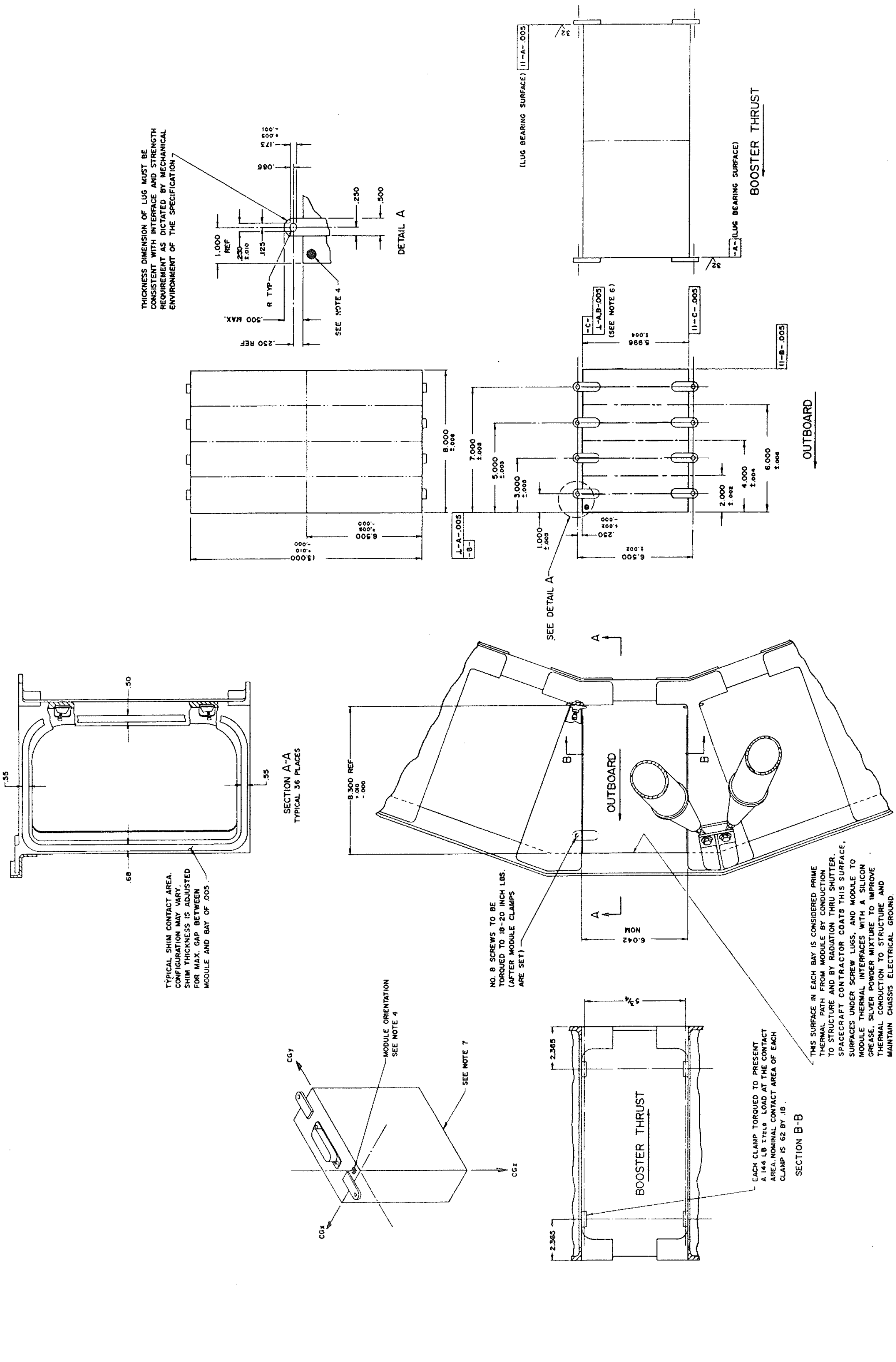
Installation Restraints (Alignment)

- SENSOR ALIGNMENT - All experiment sensors which must be aligned, or whose alignment relative to the spacecraft axes must be defined, shall incorporate alignment surfaces or indices.
- ALIGNMENT PADS - Alignment pads must be located on rigid elements of the component structure.
- COMPONENT ALIGNMENT - The spacecraft contractor will align or measure between the component alignment index and the spacecraft axes only. The relationship between the index surface and the component is the responsibility of the experimenter.
- ALIGNMENT SURFACES - Alignment surfaces need not be continuous, but must define a plane parallel or perpendicular to the component mounting face. The required geometric properties for alignment surfaces or indices are dependent on alignment tolerance:

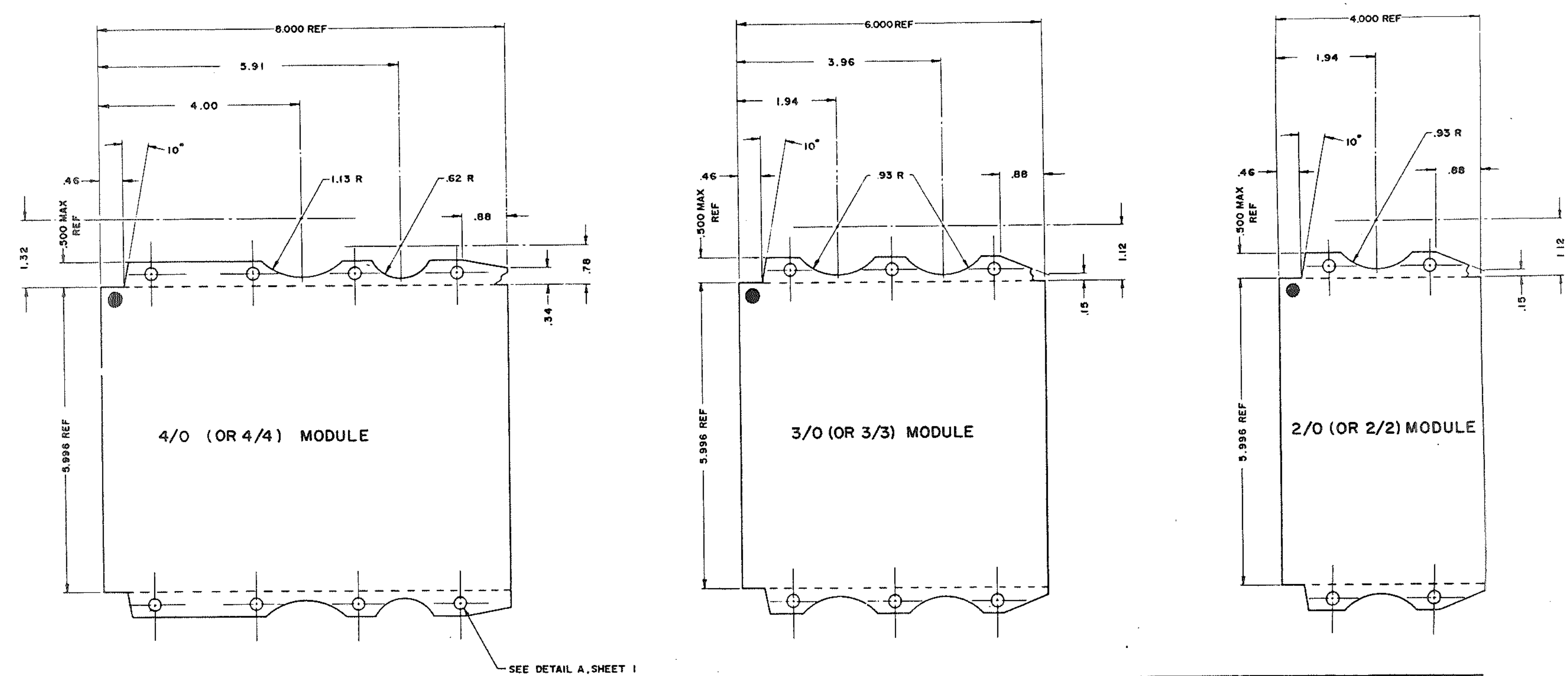
Component Alignment Limits	Surface Finish	Pad Flatness	Max. Surface Slope	Pad Size or Span (Min.)
$\pm 0.5^{\circ}$	125 min.	0.001 in.	0.005 in/ft	1.0 in.
$\pm 0.1^{\circ}$	32 min.	0.0005 in.	0.0004 in/ft	1.0 in.

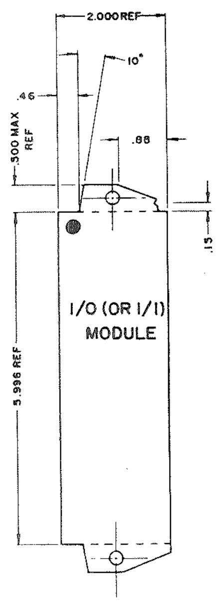
- AXES ALIGNMENT - For items requiring alignment with respect to all three axes, two mutually perpendicular alignment surfaces, or one surface with dowell pins, must be provided. For alignment with respect to two axes, one alignment surface will suffice.
- BAY-MOUNTED SENSORS - Bay mounted sensors must have alignment capability built into the component, since there is no capability for alignment of bay mounted modules. If no means of alignment are provided within the bay mounted sensors, the alignment tolerances to the mounting surfaces will be 15 minutes of arc in yaw and pitch.
- SENSOR MOUNTING - All sensors except bay mounted will be mounted orthogonal to the spacecraft axes.
- AXES IDENTIFICATION - The spacecraft axes must be identified on each sensor which is not bay mounted.

MODULAR INTERFACE DRAWING (Sheet 1 of 2)



MODULAR INTERFACE DRAWING (Sheet 2 of 2)





NOTES

1. All individual tabs must be attached to the module by at least two screws (No. 10-32) into topped inserts. For continuous tabs (requiring approval of the spacecraft manager), at least four tapped holes utilizing inserts (No. 10-32) for each 1/0 module must be provided into the tabs (two for each tab). Screws and inserts must extend completely through the module face (or through the tab if a continuous tab).

2. Phantom lines indicate sizes of modules.

3. Where power dissipation requirement is critical, an integral tab or flange design (requiring approval of the spacecraft manager) may be used (as defined by sheets 3 and 4).

4. Each module shall have an orientation marking as shown to designate the outboard surface.

5. All connectors and identification labels shall be on the surfaces with the fastening lugs.

6. Parallelism and perpendicularity the same for all module configurations.

7. Center of gravity coordinate system.

8. Unless otherwise specified: Dimensions are in inches. Tolerances on decimals = ± 0.01, ± 0.005, Tolerances on fractions = ± 1/64, Tolerances on angles = ±0.2°

THERMAL CONTROL

Thermal Control System Characteristics

THERMAL ISOLATION - The solar paddles, sensory ring, and attitude control housing sections of the spacecraft are thermally isolated from one another.

HEAT SINK - The NIMBUS sensory ring serves as a heat sink for electronics modules. Rejection of heat is accomplished by conducting heat through the bay-mounted modules to the thermal radiating surface at the sensory ring periphery.

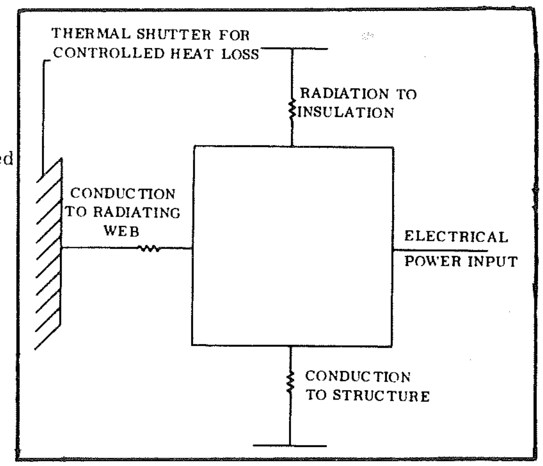
PRIMARY PATH - The primary path of heat rejection for crossbeam-mounted modules is via conduction to the sensory ring where the excess heat is radiated to space via the combined active and passive thermal control system.

SECONDARY PATH - A secondary path for crossbeam-mounted components has been used on past NIMBUS Spacecraft via use of a radiation plate, but this is contingent upon the location of few components below the crossbeam to minimize interference with the radiation of heat.

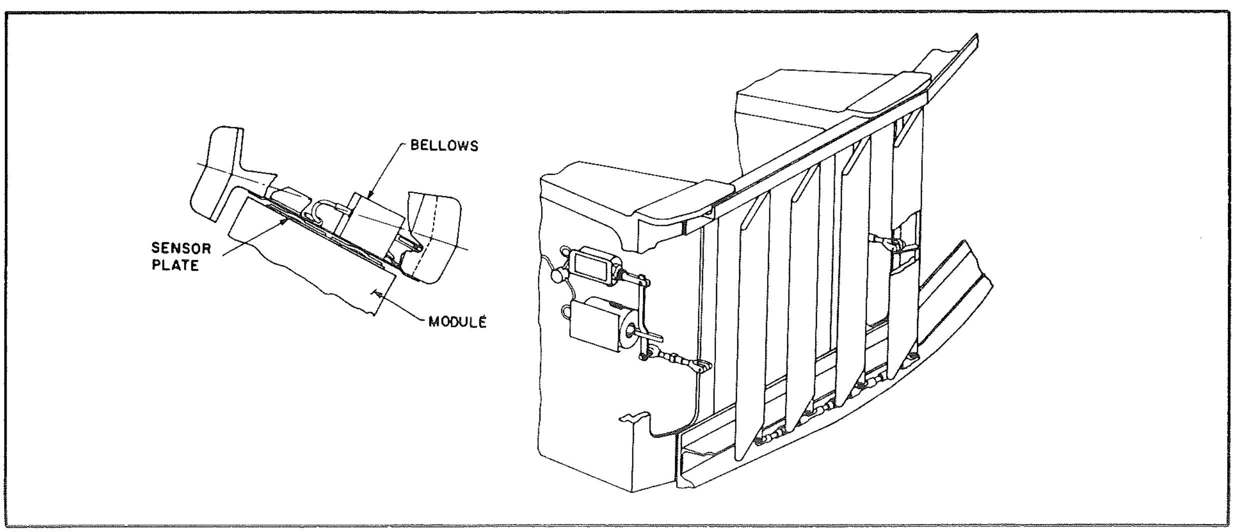
ACTIVE and PASSIVE THERMAL CONTROL - The sensory ring thermal control system utilizes both active and passive thermal control to maintain an environment in which the spacecraft subsystems and experiments can perform effectively and reliably with a minimum of degradation. A thermal balance is attained via conduction to the radiating web, conduction to the structure, and radiation to insulation (see sketch opposite).

- ACTIVE CONTROL - Active control, when required, is provided by louvers on the appropriate bays of the sensory ring. The louvers are controlled by freon-filled bellows located between sensory ring bays (see Thermal Shutter Assembly sketch below). The relative location of shutters, active controllers, and the bay-mounted module are also shown in the sketch. The active system has a fail-safe feature to return the louvers to a predetermined position in the event of failure of the bellows.

- PASSIVE CONTROL - Passive control consists of multiple layers of aluminized mylar insulation which line the panel covers to minimize the effects of solar heating and other variable environmental factors, so that more precise control can be provided by the active system.



THERMAL BALANCE SCHEMATIC



TYPICAL SHUTTER ASSEMBLY

General

EXPERIMENT THERMAL COVERS and INSULATION - Under certain conditions it may be desirable for an experimenter to provide the thermal insulation and insulation support structure. In this event, early approval by the GSFC spacecraft manager and notification of spacecraft contractor are required, and it is then the responsibility of the experimenter to negotiate a mechanical interface with the GSFC spacecraft manager and to document this interface (such documentation to include mechanical interface drawings).

RING TEMPERATURE - Tolerances in sensory ring temperature held by the thermal control system are 25±10°C.

RING-TO-MODULE TEMPERATURE DIFFERENTIAL - This differential is limited to 5°C resulting in a tolerance in module temperature of 25 (+15°C, -10°C). This differential is measured from the local structure to module mounting points and is held to this level by use of such techniques as:

- MODULE PRELOADING - The preload of 144 ±72 lb is provided by four pads (0.62 x 0.18 inches each as specified in the section on "Structural Restraints") to keep each module in good thermal contact with the outboard web. Spring loading of the module to the thermal sensing plate results in the requirement for the tight tolerance of 5.996 ±0.004 in the module width.
- MODULE COATING - Module metal mating surfaces are to be coated with silicon grease to improve the heat conduction path between modules in a bay and between outboard module surface and the thermal sensing plate.
- COMPENSATION LOAD HEATERS - Modules controlled totally or in part by passive means, and which possess operational duty cycles in which the input power is reduced to zero, may require the heat generated by a compensation load heater to maintain its temperature at a minimum of +15°C.

Thermal Constraints

MODULAR - The maximum thermal dissipation permitted within a module of a given size is given below. Special approval by the NASA GSFC spacecraft manager is required for exception to this rule, since exception will result in significant restraints upon where modules can be located in the ring and may require the removal of insulation.

Module Size	Power (Watts)
4/4	12
3/3	9
4/0, 0/4, 2/2	7
3/0, 0/3	6
2/0, 0/2, 1/1*	4
1/0, 0/1	2

*Use of 1/1 Modules To Be Avoided

NON-MODULAR - The following constraints apply to hardware which is not to be located in a bay of the sensory ring.

- PACKAGES HELD TO RING TEMPERATURE (some temperature in the 10°C to 35°C range) must not dissipate more than 3.0 watts. GSFC spacecraft manager approval is required for exception to this rule, as thermal control is made difficult by the two factors listed below. Experimenters must, however, request spacecraft manager approval to dissipate more than 3 watts if limiting the dissipation to below 3 watts results in an unsound design practice (such as bringing low level signals out of modules).
  - THE AVAILABLE SURFACE AREA for the rejection of heat is limited.
  - EARTH REFLECTED SOLAR ILLUMINATION reduces the efficiency of heat rejection by radiation.
- PACKAGES NOT HELD TO RING TEMPERATURE also require NASA GSFC spacecraft manager approval, as
  - PACKAGES HELD TO BELOW RING TEMPERATURE require cold space viewing area at the periphery of the ring. This area is at a premium as it is also used by experiments which require a cold space view for calibration.
  - PACKAGES HELD TO ABOVE RING TEMPERATURE represent localized hot spots and their existence must be known for integration purposes.

THERMAL CONTROL (Cont'd)

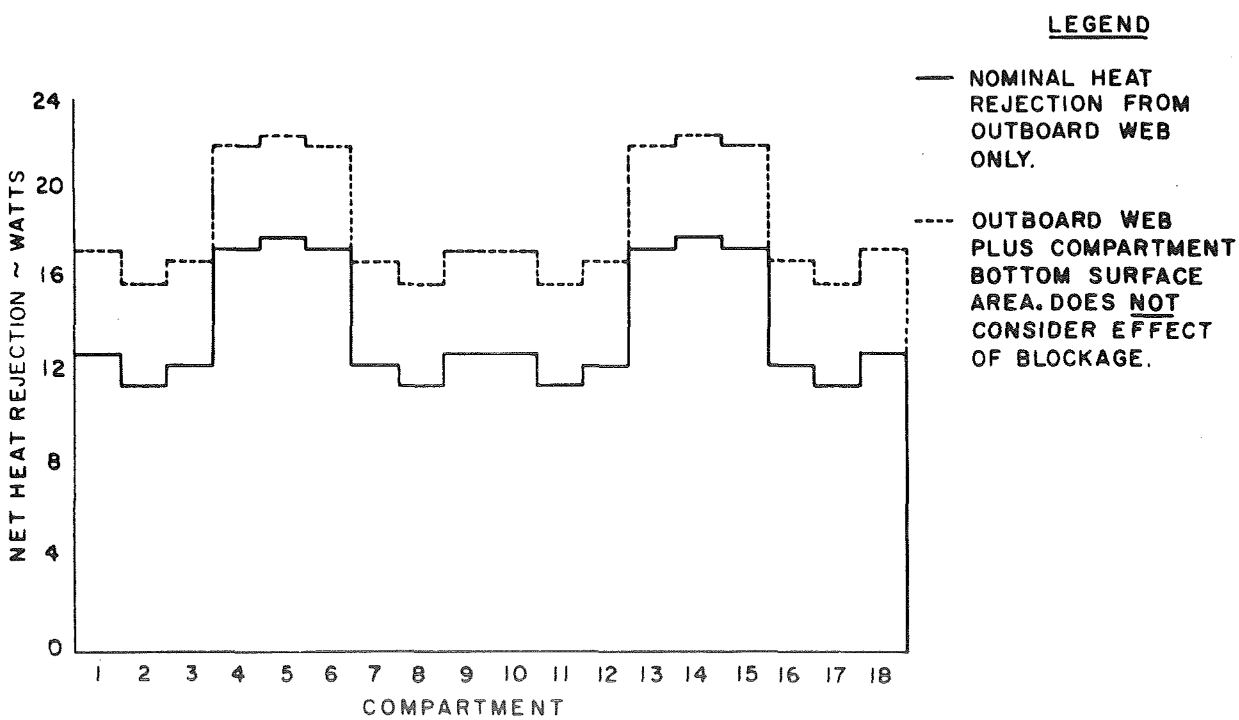
Experiment Thermal Requirements

It is imperative that the following practices be utilized in the design of experiments to ensure maximum efficiency of heat rejection.

- HIGH HEAT SOURCES must be located as close as possible to the outboard facing surface of the module.
- AMPLE SURFACE AREA must be used so that good thermal contact occurs between the base of components which are heat sources and the inside surface of the module.
- HEAT SINKING OF POWER TRANSISTORS - When heat sinking power transistors, use Berlox washer and Apiezon grease combination, or an equivalent.

NOTE

Do not use Berlox washer and indium foil as indium foil will cold-flow through cracks in the Berlox washer and short circuit the transistor to its mount.



SENSORY SUBSYSTEM HEAT REJECTION CHARACTERISTICS

Techniques Available During System Integration

The following techniques are utilized by the spacecraft contractor during integration to ensure adequate rejection of heat by the module. These techniques are only useful if experiment module thermal design is adequate, and it is possible to use only a few of these techniques in any given situation.

- MODULE LOCATION - Modules will be located in accordance with the heat rejection capabilities of the individual bays.
- INSULATION REMOVAL - Local removal of insulation under a given bay can be used to increase the heat rejection capability of the bay by up 4 watts. This procedure can not be relied upon to solve serious thermal problems, since it may be necessary to perform this procedure initially.
- CROSSBEAM-MOUNTED COMPONENTS - Low heat dissipation components are selected to the extent possible for mounting in the crossbeam area.
- OUT-OF-PHASE - Components will be located in the crossbeam and sensory ring so that the duty cycles for components within an individual bay are out-of-phase to the extent possible.
- IF NOT IN PREFERRED LOCATION - When high heat dissipating modules with limited periods of operation cannot be accommodated in the preferred location, they will be mounted in bays facing away from the sun during the periods of operation.
- LOCATION OF MODULES WITHIN A BAY - In bays where several modules are to be located, these modules will be arranged in order of heat dissipation rates with the modules with the highest heat dissipation rate located closest to the outer periphery of the sensory ring.
- HEAT REJECTION MARGIN WITHIN A BAY - To the extent possible, only 70 to 80% of the heat rejection capacity of a bay will be utilized to provide margin for subsystem degradation and changes in duty cycle. The heat rejection capability of the different bays is shown in the sketch on the opposite page.
- CROSSBEAM RADIATOR PLATE - Crossbeam mounted components may be heat sinked to a radiator plate to increase the heat rejection capability of the crossbeam area.
- THERMAL COMPENSATION LOADS - Electrical heaters are mounted on modules, structure, and the crossbeam radiation plate to the extent required. They are turned on as required to compensate for components which are off in maintaining the temperature tolerances of the spacecraft.

Coatings

D4D and PV100 coatings are utilized where required to aid in thermal control by controlling the absorptive and emissive properties of surfaces.

a. PV100 (High ϵ and Low α)

PV100 is painted on part of the structure and components to radiate heat. Examples of the use of PV100 are:

1. A circular area of outboard surfaces of outboard modules is coated with PV100 for radiation through the shutters.
2. Where local removal of insulation is used, PV100 is put on the bottom surfaces of bottom loaded modules to increase the heat rejection capability of a bay.
3. The crossbeam radiator plate is coated with PV100.

b. D4D (Low α and ϵ)

D4D is used to desensitize a component to its environment by attaining low absorptive and radiative properties for the component surface.

ATTITUDE CONTROL SYSTEM

The Attitude Control System employs three-axis active control to maintain the spacecraft orientation with respect to the local vertical and the orbit plane. It also provides independent single-axis control of each solar paddle for sun tracking.

Description

- LOCATION - The Attitude Control System is located in the upper housing to which the oriented solar array drive shafts are attached. It is structurally tied to the sensory ring by a truss consisting of six members knee-mounted in sockets which are laterally adjustable for alignment purposes.
- TORQUERS - Reaction wheels are used for fine control and residual momentum storage, and mass expulsion provides net momentum control and large control torques when required.
- SENSORS - Attitude error sensing is provided by conical scan earth sensors, sun sensors, and rate gyros.

Alternate Modes

The Attitude Control System also contains an extensible rod with tip mass to utilize gravity gradient torques in two modes which do not require pneumatics:

- Active control augmented by gravity gradient torquing
- Active control augmented by gravity gradient torquing and a pitch momentum bias

Attitude Control Restraints

The following rules must be adhered to to ensure that the capabilities of the Attitude Control System are not exceeded:

- When subsystems have rotating parts, the spin axis of the rotating part must be aligned to the spacecraft pitch axis to the maximum extent practical.
- The GSFC spacecraft manager must approve the use of rotating members with a residual angular momentum greater than 0.005 ft-lb-sec. The orientation of this momentum with respect to the spacecraft axes must be specified.
- The GSFC spacecraft manager must approve the use of translating parts, or rotating parts whose center of mass is not at the center of rotation and which have a mass displacement of greater than 4.36 lb-inch during separation and 10.0 lb-inch following acquisition.

Position Accuracy

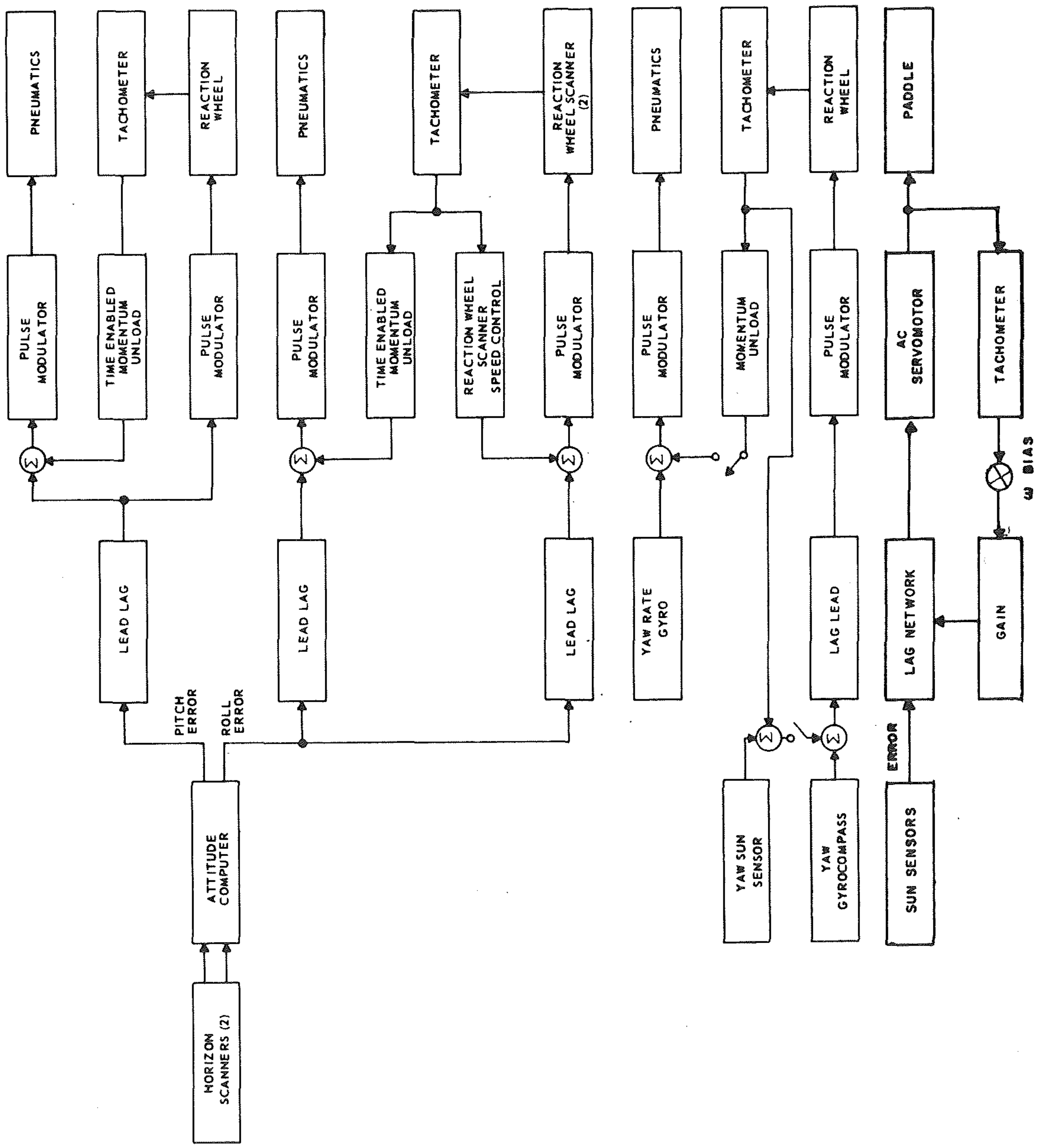
Pointing accuracy with respect to the reference orientation is within 1 degree for each of the three body axes.

Spacecraft Rates

Spacecraft rates are specified to be less than 0.05 degree/sec about roll and yaw after initial acquisition of the attitude reference. The angular rate about pitch will be 0.055 $\pm$ 0.05 degree/sec (0.055 degree/sec is the orbital rate).

ATTITUDE CONTROL SYSTEM

(Primary Mode)



WEIGHT, POWER and SIZE CONTROL

IT IS IMPERATIVE that control of experiment WEIGHT, POWER, and SIZE be incorporated into the design of experiments. Once particular locations on the spacecraft have been selected for experiment hardware, a change in any of these parameters would seriously degrade the experiment environment and/or increase the difficulty of spacecraft integration.

Control of Experiment Weight

Control of experiment weight is needed so that:

- THE STRUCTURAL CAPABILITY of the hardware required to support the experiment and of the spacecraft structure is not exceeded.
- THE SPACECRAFT MASS PROPERTIES can be known and held to acceptable values.

Control of Experiment Power

Control of experiment power is required so that:

- THERMAL CONTROL can be maintained. An increase in experiment power could cause the heat rejection capability of a bay to be exceeded or result in a sensor package operating too hot since the heat rejection capability below the ring is limited.
- POWER SYSTEM CAPABILITIES are not exceeded and operational power profiles can be generated and maintained.

Control of Experiment Hardware Size

Control of experiment hardware size must be maintained so that:

- THERMAL CONTROL to the specified tolerances can be achieved. The radiating area to heat dissipated ratio is a parameter which determines the experiment operating temperature. Therefore, all dimensions of an experiment package must be controlled.
- MODULAR PACKAGES can be integrated into the sensory ring. A change in a modular package size could result in the change of several bay assignments, a degradation in spacecraft mass properties or thermal control, or in the sensory ring capacity being exceeded.
- SENSOR PACKAGES can be located below the sensory ring. Since selecting a sensor package location is a complicated function of the size, heat dissipation, field-of-view, and various other factors, a change in size could require a sensory ring redesign.

Program Impact

The primary purposes of these controls are obvious but important enough to state:

- ACCURACY OF INITIAL ESTIMATES - Careful, realistic estimates can result in significant program savings and minimize spacecraft changes.
- PROMPT REPORTING OF CHANGES - The earlier a change is reported in the system design cycle - the lesser the effect.
- TIMELY REPORTING - Consistent compliance with monthly reporting requirements will allow the current spacecraft design status to be maintained.
- EARLY AVAILABILITY OF DESIGN INFORMATION - Spacecraft design "freeze" is dependent on methodical and timely progress from ESTIMATED-TO-COMPUTED-TO-ACTUAL values. The lack of data on one experiment could impact the entire Program.

Procedures

- ASSIGNMENTS - WEIGHT, POWER, and SIZE assignments will be mutually agreed upon by the GSFC spacecraft manager and the experimenter. Assignments will be made on a package by package basis, rather than a single value being assigned to an experiment and the breakdown left unspecified.

- ASSIGNMENT COORDINATION - Assignments must be coordinated with the spacecraft contractor who records, audits, and reports the weight, power, and size assignments for the entire spacecraft.

- MONTHLY REPORTS shall be submitted to the GSFC spacecraft manager. The monthly reports must state the original weight, power, and size and the values of these parameters for the current and previous months. The precentage of weight, power, and size which is ESTIMATED, COMPUTED, and ACTUAL must be given for each reported value.

- CONTROL MEASURES - The experimenter shall perform the following weight, power, and size control measures during experiment design:

  - PREPARE A PARTS LIST for each experiment package (parts lists are a requirement for design development). Assign a weight, power consumption/dissipation, and size for each line item indicating by E, C, or A whether the value is ESTIMATED, COMPUTED, or ACTUAL.

  - EVALUATE THE DESIRABILITY of incorporating the alternatives indicated by analysis. (Mechanical and electrical requirements for functional adequacy and reliability, and for surviving environmental tests, must not be compromised.)

  - INCORPORATE WEIGHT, POWER, AND SIZE SAVING ACTION. Keep the weight, power, and size listing up to date as work progresses on completing the design. Convert reported values to actual values as soon as possible.

  - INCLUDE STATUS IN MONTHLY TECHNICAL REPORTS - Include changes since last report. Justify any increases or contemplated design changes which could affect weight, power, and size. Identify the precentage of these parameters which is estimated, computed, or actual.

- DESIGN REVIEWS - The GSFC spacecraft manager will conduct design reviews to assure that the design, development, and fabrication program will effectively accomplish control of experiment weight, size, and power.

- APPROVALS - Special approval by the NASA GSFC spacecraft manager is required for:

  - ANY INCREASE in the weight, power, or size assignment over the original negotiated value. Documented justification for the increase must accompany any requests for increased assignments.

  - ANY MODULAR PACKAGE which exceeds the following average power dissipation when dissipation is averaged over any 15 minute period:

Modular Size	Power Dissipation (Watts)
4/4	12
3/3	9
4/0, 0/4, 2/2	7
3/0, 0/3	6
2/0, 0/2, 1/1*	4
1/0, 0/1	2

*1/1 Modular Size To Be Avoided

  - ANY NON-MODULAR PACKAGE (sensor) in which average power dissipation exceeds 3 watts or which is to be held to some temperature other than a temperature in the 10 to 40°C range. The experimenter must consult the GSFC spacecraft manager immediately in cases where limiting the dissipation to below 3 watts results in an unsound design practice (such as bringing low level signals out of modules).

- MASS PROPERTIES must be reported in the monthly weight, power, and size reports and in the interface agreement.

  - COMPONENT CENTER OF GRAVITY - The location of the center of gravity of all packages must be defined within ± 0.1 inch.

  - COMPONENT MOMENTS OF INERTIA - Not required.

Section III

ELECTRICAL REQUIREMENTS

	Page No.
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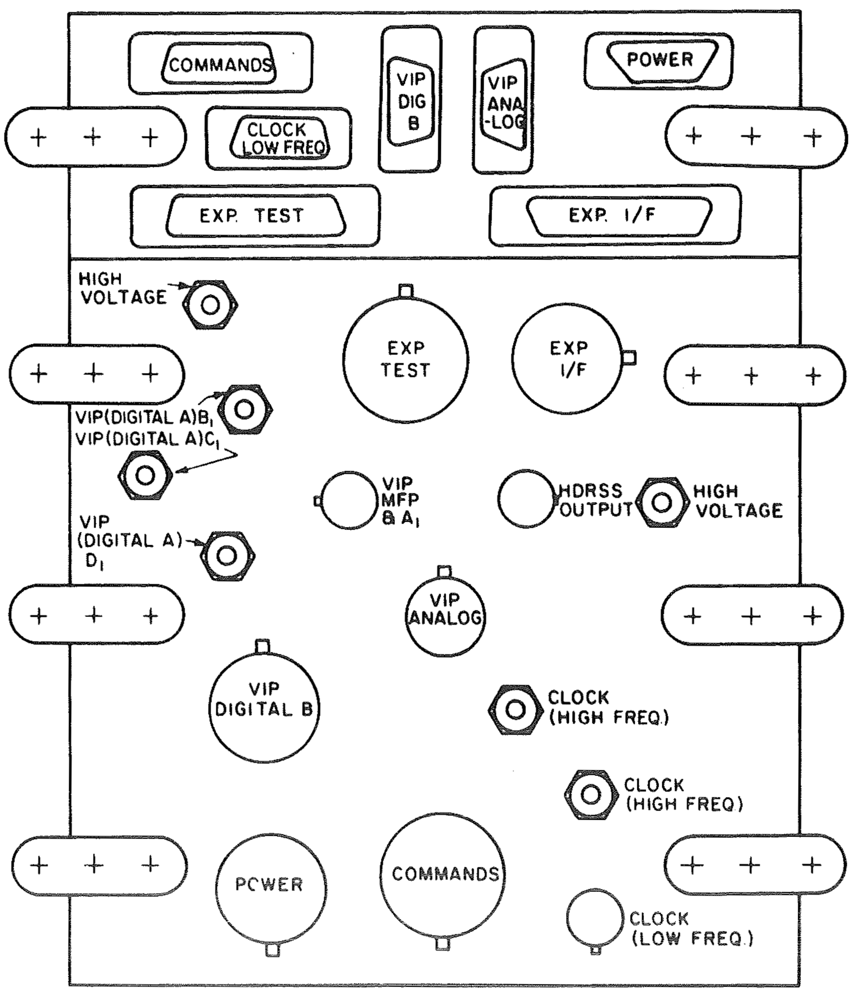
Clock.	3-6
Commands	3-8

ELECTRICAL INTERFACE and CONNECTORS

Typical NIMBUS Module

A permissible connector arrangement for a NIMBUS module is illustrated below. Note that connectors are mounted only on the mounting tab faces of the module. Whether the module is bay-mounted or not, separate connectors must be provided for:

- Power
- All low frequency (50 kHz or less) clock signals
- Each high frequency (100 kHz or greater) clock signal
- Commands
- VIP analog telemetry
- VIP digital "B" telemetry
- Each VIP digital "A" telemetry pulse line
- VIP timing
- HDRSS output
- Each high voltage (above 250 volts)



MODULE CONNECTOR ARRANGEMENT

ELECTRICAL INTERFACE and CONNECTORS (Cont'd)

Selection of Connectors

The following rules must be followed in the selection of connectors:

- UNPRESSURIZED MODULES

  Connectors, multicontact, on all unpressurized modules except those for high voltages, high frequencies, and analog and digital B telemetry must be either of the following:

  MIL-C-38999 - Bendix JTN type
  MIL-C-24308 - Cannon D subminiature type; crimp or solder contact type

  Although the rectangular shell type MIL-C-24308 connectors may have to be used for 1/0 modules so that enough connectors can be mounted on the long and narrow module surface, the use of the MIL-C-38999 connector is preferred for all module sizes.

  The MIL-C-38999 connectors are of the crimp contact type, and connector shells shall be noncadmium plate.

  The MIL-C-24308 connectors are of the solder or crimp contact type. Connectors shall be in accordance with Class N of MIL-C-24308, except the connector shells shall be gold plated in accordance with MIL-G-45204, Type II, Class 2, over copper plate, 0.000050 inches thick, per MIL-C-14550. Solder type contacts shall be plated 0.000050 inches gold per MIL-G-45204, over 0.000200 inches silver per QQ-S-365 over copper plate 0.000030 inches per MIL-C-14550. Reference MIL-C-24308/5, /6, /7, and /8.

  NOTES

  1. The solder contact and crimp contact MIL-C-24308 connectors are interchangeable and intermateable.
  2. Recommended crimping tools

MS Part No.	Buchanan Electrical Products Co. Part No.
MS 3198 (with MS 3198-5 locator)	Tool No. 612596 (with 613533 locator)
MS 3191-1	Tool No. 10692 (with 11697-1 locator)

- HIGH VOLTAGES - Multicontact connectors must not be used for voltages in excess of 250 volts ac peak-to peak or 300 volts dc. (Use of voltages in excess of 250 volts requires approval by the GSFC spacecraft manager.)

- HIGH VOLTAGE CONNECTORS - Use separate high voltage rated connectors for voltages in excess of 300 volts. The connector selected must be approved by the GSFC spacecraft manager.

- HIGH CLOCK FREQUENCY CONNECTORS - These connectors (100 kHz or greater) must be twin-axial type. Individual connectors must be used for each high clock frequency. The connector used must be approved by the GSFC spacecraft manager.

- ANALOG OR DIGITAL B TELEMETRY - Special connectors containing filter type contacts of the MIL-C-26482 type such as Bendix series FJT must be used. They must provide an attenuation of at least 50 db from 100 to 10,000 MHz at temperatures from -55 to +125°C.

- PIN SIZE - Use of connectors with pins as small as No. 22 (available in the MIL-C-38999 series) is permissible for applications other than power (high voltages require special connectors). But the No. 22 pins must be able to accommodate a No. 20 wire, which may be required to reduce losses in the spacecraft harness. No. 20 pins or larger are required for power, depending on the subsystem current.

- PRESSURIZED MODULES

  Pressurized modules must be fabricated using hermetically sealed connector receptacles as follows:

  - STANDARD CONNECTORS - For standard connectors (no-high density), use connector types mateable with specification MIL-C-26482, such as the Deutsch DTK-H or Bendix PTH series.

  - HIGH DENSITY CONFIGURATIONS - For high density configurations, use connectors meeting Class Y of specification MIL-C-38999, such as the Bendix JTNH series.

  - COAXIAL CONNECTORS - Coaxial connectors shall be the TNC and miniature types in accordance with MIL-C-39012.

Use and Location of Connectors

The use and location of connectors must be in accordance with the following:

- ALLOW SPACE between connectors (at least 1/2-inch) for fingers, clips, etc.
- CONGESTED AREAS
  - USE DIFFERENT SIZE CONNECTORS in congested areas wherever possible.
  - ALTERNATE KEYING ARRANGEMENTS must be used if two or more circular connectors of the same size must be used in a congested area.
  - INVERT ONE of the connectors if two or more MIL-C-24308 connectors of the same size must be used in a congested area.
  - HIGH DENSITY CONNECTORS are available in the MIL-C-38999 series.
- FULL DEPTH MODULES - Each module of size 1/1 (to be avoided) 2/2, 3/3, and 4/4 should have all its connectors mounted on only one of the module connector mounting surfaces. In sensory ring integration, the module will be oriented so that the connector surface is on top to ease harnessing problems.
- IF LARGE CONNECTORS ARE USED, locate them in the middle of the module face.
- POWER CONNECTORS
  - AT LEAST TWO CONNECTOR PINS must be wired for each spacecraft power input and power return.
  - FOR NO. 20 CONNECTOR PINS AND WIRES, additional pins must be used if the current exceeds 3.0 amps.
  - FOR NO. 16 CONNECTOR PINS AND WIRES, additional pins must be used if the current exceeds 5.5 amps.
- MALE CONNECTORS must be used for input signals or input power, and female connectors for output data or output power.
- SPARE PINS - A minimum of 10 percent spare pins per connector shall be provided at the beginning of the program. These spare pins must be located on the outer periphery of the connector.
- SHIELDED WIRES brought to a connector shall have the shield connection on a contact adjacent to the signal contact.
- TWISTED PAIR - Wires to be twisted must be on adjacent pins.
- LOW LEVEL SIGNALS - (full-scale level of less than one volt) must not be brought out to connectors (amplify first to at least one volt amplitude).
- GROUND PINS - Each connector (except coax) shall have two signal ground and one chassis ground pin assignment. These pins are required for proper grounding of shields and for providing a signal reference on each connector.
  - BOTH SIGNAL GROUND PINS must be wired to the signal ground point in the experiment circuitry. Two pins are required so that parallel wires can be run in the spacecraft harness to reduce the harness impedance and for redundancy.
  - THE CHASSIS GROUND PIN must be wired to the module chassis for use by the spacecraft contractor in the grounding of spacecraft harness shields. The impedance from the connector pin to the chassis must be less than 5 milliohms.
- HIGHER VOLTAGE CONTACTS shall be located farthest from the ground contacts on the same connector.
- KEEP SEPARATE - Connectors which interface with other spacecraft subsystems must be located separately on the module face from intraexperiment or intrasubsystem wiring.
- ALL TEST POINTS must be buffered to prevent damage due to external short circuits.
- EXPOSED TERMINAL COATING - All exposed terminals on prototype and flight models must be conformal coated.

POWER

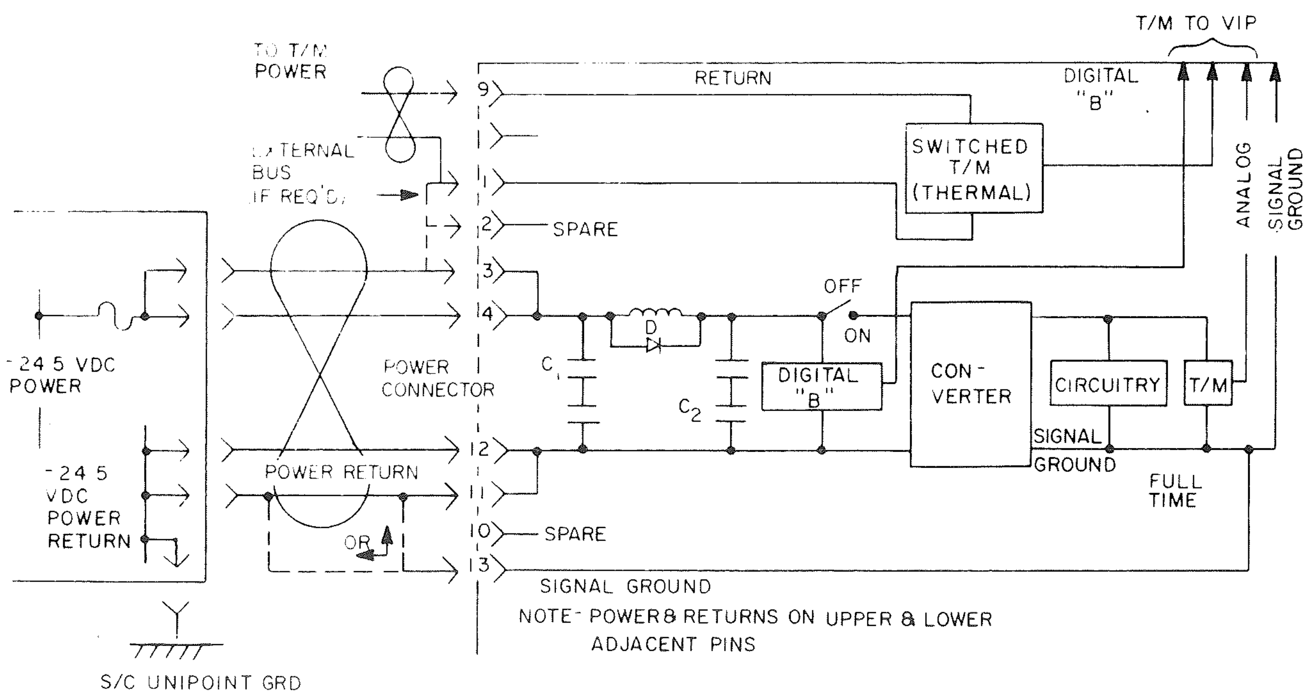
Description

- The available dc power is regulated -24.5 ± 0.5 vdc and unregulated -26.5 to -37.5 vdc. Experiments should use the regulated bus for required power. However, unregulated power may be utilized upon approval of the GSFC spacecraft manager (motors, solenoids, etc. should use unregulated power).
- DC-DC Converters must be used to isolate the signal ground from the power return.
- All power lines and power returns must be wired to a connector separate from other electrical signals.

Power Distribution Restraints

Power distribution restraints are explained below and illustrated in the sketch at the bottom of the page. The pin geometry of a 15-pin Cannon D connector is assumed in the illustration, but the restraints remain applicable if the Bendix JT connectors are used.

- ADJACENT PINS - Power and power returns must be wired to upper and lower adjacent pins on the connector in all cases (Pins 1 and 9, 3 and 11, and 4 and 12 are adjacent on a 15 pin connector).
- WIRE AT LEAST TWO PINS for input power (pins 3 and 4) and for the power return (pins 11 and 12). Currents greater than 3 amps will require additional pins.
- FOR TELEMETRY MONITORS switched on and off, wire power and power returns to pins separate from the main power inputs (pins 1 and 9, rather than 3, 4, 11 and 12).
- PLACE SPARE PINS adjacent to pins where additional bussing is expected or desired (pins 2 and 10).
- TURN ON TRANSIENT - Place a filter capacitor ahead of the switch to eliminate turn-on transients (capacitor C_2).
- TURN OFF TRANSIENT - If chokes are used on the input lines, utilize a quenching diode to eliminate turn-off transients (diode D)..
- WIRE SIGNAL GROUND to pins separate from the return on the power and VIP connectors (pin 13).



POWER DISTRIBUTION RESTRAINTS

Power Regulation Restraints

- STEP LOADS - The experiment must be so designed that a step load of greater than 2 amperes with a rise time of less than 10 microseconds does not occur. If step loads of greater than 2 amperes cannot be avoided, the rise time of the step load must be increased to be in excess of the minimum values of the graph shown below. This rule is mandatory as non-compliance will result in loss of regulation of the regulated bus.

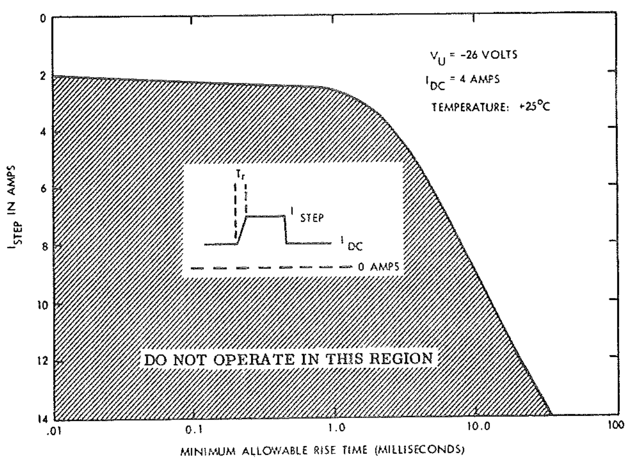
- TRANSIENT DURING REGULATOR MALFUNCTION - The experiment must be capable of surviving a regulator malfunction (regulated bus drops to -18 volts or rises to -39 volts for 40 milliseconds during the time the redundant regulator is switched).

- POWER LINE NOISE - The experiment must operate within specifications with noise on the power line of as much as 0.25 volts peak-to-peak at any and all frequencies.

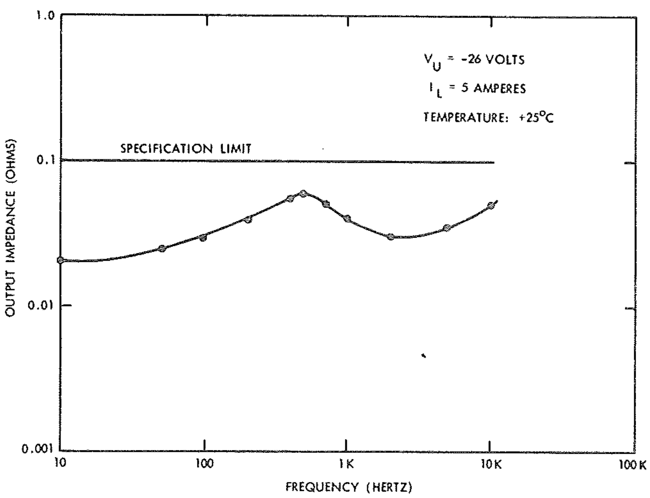
- FEEDBACK RIPPLE OR NOISE - The peak-to-peak value of ripple or noise voltage fed back on the regulated bus is not to exceed 25 millivolts. The peak-to-peak value of ripple or noise current fed back on the regulated bus must not exceed 10% of the steady-state current. Adherance to both of these specifications is mandatory.

- SURVIVAL OF VOLTAGE VARIATION - The experiment must be capable of surviving voltage levels of between -20 and -34.5 vdc continuously at 35°C with no resultant degradation in experiment life.

- OUTPUT IMPEDANCE VERSUS FREQUENCY - A plot of the output impedance of the power supply versus frequency is shown below as an aid in designing filters and simulating the spacecraft power supply during tests.



RISE TIME VS. PEAK CURRENT



POWER SUPPLY OUTPUT IMPEDANCE AS A FUNCTION OF FREQUENCY

Experiment Fusing

Experiments shall not supply fusing. All fusing (and subsystem current monitoring) will be accomplished in the spacecraft electrical system. Fusing will be performed in accordance with the following:

- FUSE SIZE will be three times the steady-state current or greater.

- MINIMIZE NUMBER OF FUSE SIZES - Only 1 amp, 5 amp, and parallel combinations of these fuses will be used.

- SHORT CIRCUIT - Fuses are mainly used for short circuit protection.

- CRITICAL FUNCTIONS (attitude control system, command clock, etc.) will not be fused, as their loss would terminate the mission.

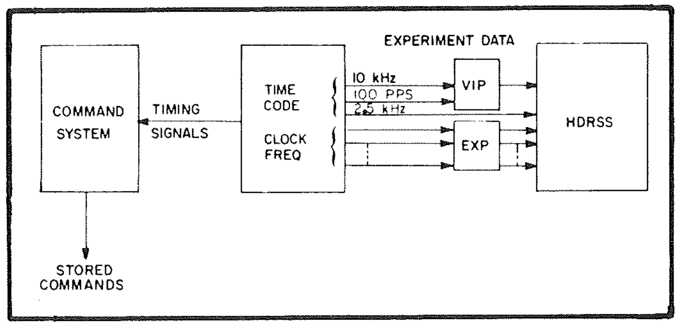
CLOCK

Clock Characteristics

THE CLOCK PROVIDES an accurate time reference in seconds, minutes, hours, and days, and provides stable frequencies for use by experiments and subsystems (see sketch below).

ACCURATE TIME REFERENCE is usable by the experiment in two ways:

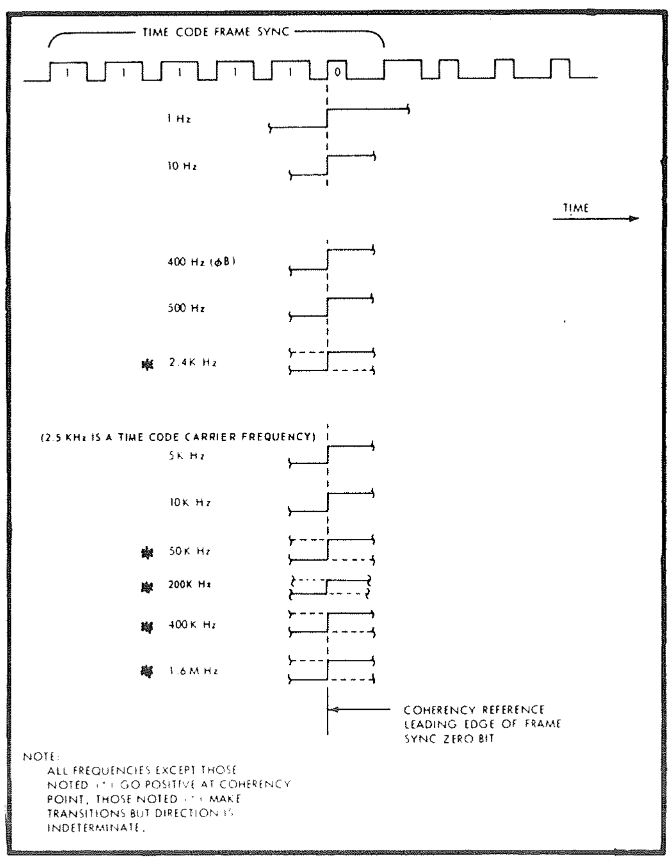
- STORED COMMANDS - The clock provides a reference for the command system so that a limited number of commands can be executed at a predetermined time.
- TIME IDENTIFICATION - Spacecraft time is provided at 100 pps to the VIP for placement in the VIP matrix, providing correlation of all VIP data. This time code also modulates 10 kHz and is transmitted to ground by the VIP beacon. It also modulates 2.5 kHz and is put on one of the HDRSS channels for correlation of all data stored on HDRSS and for flutter/wow information. The time reference provided by the clock is the NASA 36-bit time code (reference IRIG Document 104-59).



CLOCK FUNCTIONS

FREQUENCIES PROVIDED BY THE CLOCK are 1, 10 and 500 Hz; 2.4, 5, 10, and 50 kHz; and 0.2, 0.4, and 1.6 MHz. In addition, two phase clock signals (Phase B leads Phase A by 90 degrees) are provided at 100 and 400 Hz, and two phase motor drive signals are provided at 400 Hz. All clock frequencies are coherent square waves; the coherency is defined in the coherency chart →. Only these frequencies are to be used by experiments, except where frequency conversion is performed within the experiment hardware. The maximum allocation of a particular frequency signal to any one experiement is one. If many clock frequencies are desired for an experiment, the experiment must include frequency conversion logic to generate these signals from a single clock signal rather than the experiment contractor requesting that many clock signals be allocated.

REQUESTS FOR CLOCK FREQUENCY ALLOCATIONS must be made to both the GSFC spacecraft manager and the spacecraft contractor. Requests must be updated as soon as new requirements arise.

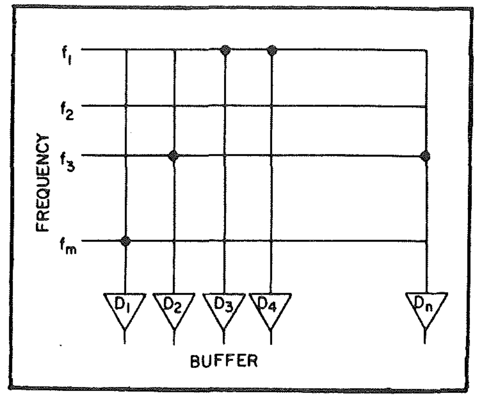


COHERENCY CHART, NIMBUS CLOCK

Clock Interface

TRANSFORMER COUPLING is to be used in the experiment for the clock signal load. NASA GSFC spacecraft manager approval of all clock interface cricuitry is required.

BUFFER AMPLIFIERS - A total of 45 buffer amplifiers are available for use with any combination of the available positive going clock signals, except 400 kHz, 1.6 MHz, and the motor drives (see sketch opposite). 10 Buffer amplifiers are available for negative going clock signals. Only one load is to be driven by a buffer amplifier. Output impedance of the positive buffer amplifiers is 1730 ± 200 ohms. Output impedance of the negative buffer amplifiers is 1730 ± 90 ohms.



BUFFER AMPLIFIERS AVAILABLE FOR USE WITH CLOCK FREQUENCIES

POSITIVE GOING CLOCK SIGNALS are to be utilized by the experiments. The negative buffer amplifiers are reserved for subsystems which are carryovers from previous NIMBUS Spacecraft. The voltage swing for the positive going clock signals is from 0.2 (+0.10 -0.15) to 5.25 ± 0.75 volts for the no-load condition and from 0.2 (+0.10 -0.15) to 1.3 (+0.3 -0.2) volts for a 600-ohm load at all frequencies except 400 kHz, 1.6 MHz, and the 400 Hz motor drive signals.

400 kHz and 1.6 Mhz - Six amplifiers, each with an output voltage greater than 1 volt P-P into a 78 ohm ± 10% load, provide outputs of 1.6 MHz and 400 kHz. Use of any of these outputs requires early GSFC spacecraft manager approval (as few outputs are available).

FOR THE 400 Hz TWO PHASE MOTOR DRIVE SIGNALS, each phase has five outputs with a voltage swing from -1.5 ± 1.0 to -23.0 ±1.5 volts with no load applied. The source impedance of each output line is 275 ± 25 ohms. Each output can drive a capacitive-coupled load of 1000 ohms or greater, or a direct-coupled load of 2000 ohms or greater returned to either ground or the -24.5 volt regulated supply. Phase B leads Phase A by 90 degrees. Use of the motor drive signals requires early GSFC spacecraft manager approval, as only a few of the outputs are available for experiments.

GROUNDING - The isolation transformers are not to be grounded to either the power return or chassis ground, but rather to a separate pin and lead is to be provided for connecting to the low level signal ground at the spacecraft clock.

CLOCK NOISE - Clock input circuits must be designed to operate within specifications with the clock input signal degraded to the following:

- A 20 db clock signal to peak noise ratioat all frequencies in the dc to MHz range.

CONNECTORS AND WIRING

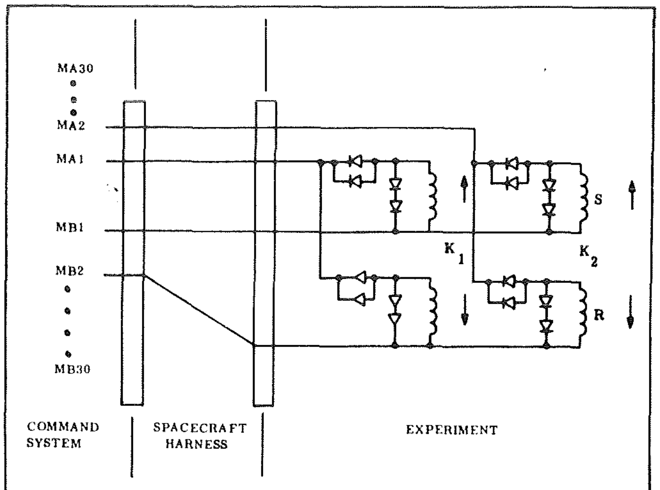
- LOW FREQUENCIES - A single Bendix JT or Cannon D connector must be used for all clock frequencies 50 kHz or less. The clock signal returns must be brought to separate pins on the connector, and are not to be connected to the power supply return.

- HIGH FREQUENCIES - Individual Twin-Ax connectors must beused for all clock signals 200 kHz or greater.

COMMANDS

Command Subsystem Characteristics

MATRIX ARRANGEMENT OF COMMANDS - The Command Subsystem provides a total of 512 commands for use by the spacecraft subsystems and experiments in the form of a 16 x 32 matrix. Any 30 of these commands can be stored at the same time for delayed execution. (See matrix sketch on this page and functional diagram on next page.)



COMMAND MATRIX

BACKUP UNENCODED COMMAND SYSTEM - This system provides up to 12 commands for spacecraft subsystems which are available in case of a command system failure to initiate critical commanded functions. These commands are typically reserved for spacecraft subsystems rather than experiments, and their use requires GSFC spacecraft manager approval.

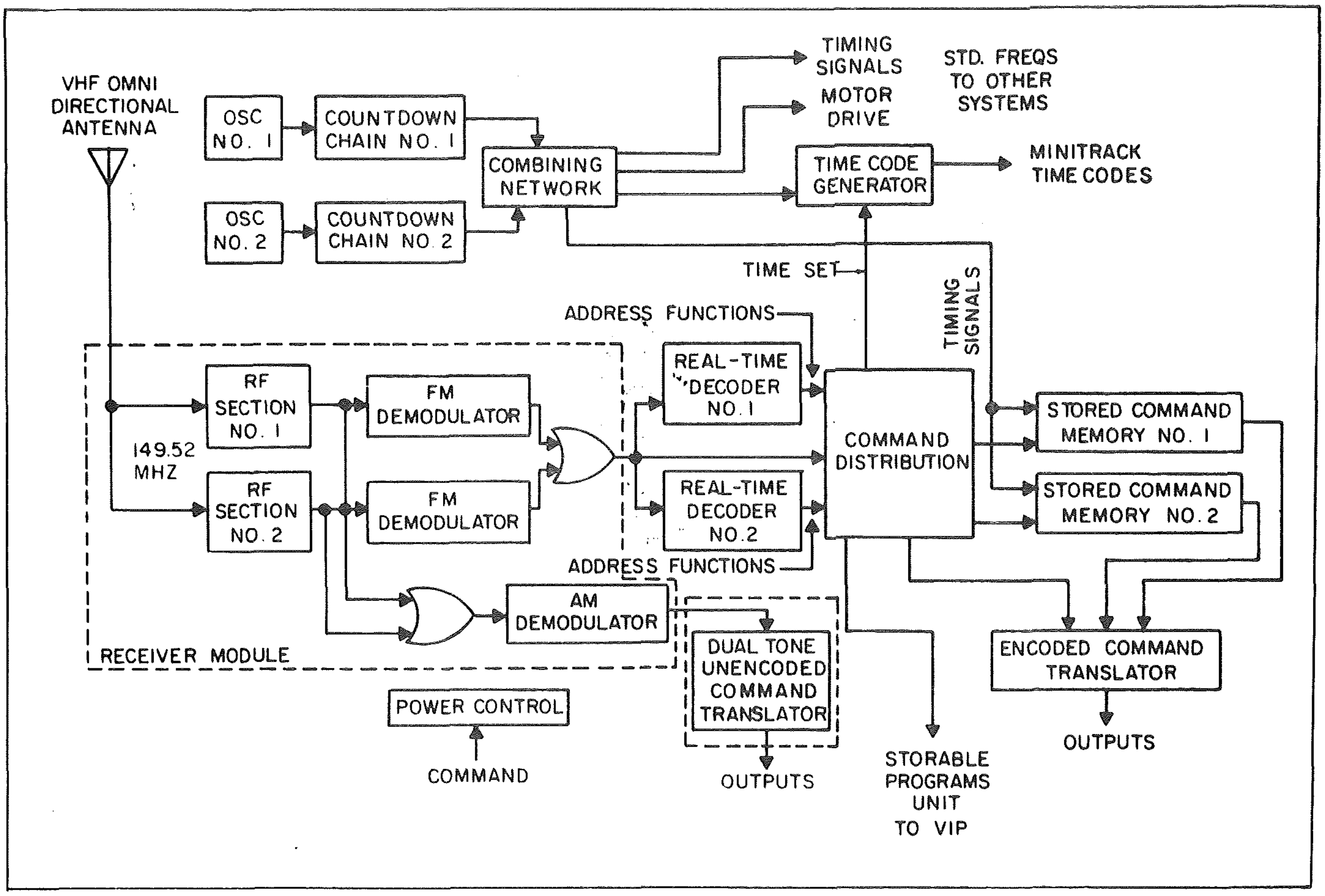
MA AND MB LINES - 16 MA and 32 MB lines provide the matrix arrangement, as one MA line and one MB line provide the input signal for each user relay. The MA line and the MB line associated with a command relay must be pulsed simultaneoulsy for the relay to be energized.

STORED COMMANDS - Stored commands are sent to the spacecraft via the RF link (as are non-stored commands) and the command module sends out the command pulses at the desired time via use of the spacecraft clock as the time reference. Stored commands can also be recycled -- the same command pulses sent out repeatedly with a preset interval between pulses. The maximum time to first execution of a command is approximately 18 hours, and the maximum recycle time is approximately 9 hours. Command timing is in increments of one second. (Early approval by the spacecraft manager is required for the use of stored commands.)

COMMAND PULSE DURATION (40 ± 5 milliseconds) - A command relay is energized by the simultaneous occurrence of a negative going voltage pulse on its MA line and a positive going pulse on its MB line.

REQUESTS FOR COMMANDS - Requests must be submitted to the spacecraft manager and spacecraft contractor and updated as soon as new requirements are determined. Requests for redundant commands must be clearly noted. The following information must accompany command requests:

1. Time of occurrence of command
2. Maximum command rate (approximately 2 real time commands can be executed each second)
3. Redundant or non-redundant
4. Will command ever have to be stored



COMMAND/CLOCK SUBSYSTEM

Mandatory Load for Command Pulse

The only permissible loads for spacecraft MA & MB commands are the relays listed below. The characteristics of the relays required for a 12 vdc coil (up to 2 amps and from 2-to-10 amps) also are listed below (the 12 volt characteristics are relevant because the command pulse can be degraded as low as 12 volts at the relay):

Switched Current (amp)	Relay S/N*	Type Relay	Maximum Operating Time (millisec)	Maximum Release Time (millisec)	Coil Resistance (ohms ±10%)
Up to 2	GE (3SAM)	Latching	5	-	300
Up to 2	GE (3SAF)	Holding	5	4	125
Up to 2	F (G13)	Holding	5	4	125
Up to 2	PB (FC)	Holding	3.5	3	210
Up to 2	PB (FL)	Latching	3.5	-	230
Up to 2	PB (SC)	Holding	3	2.5	135
Up to 2	PB (SL)	Latching	3	-	160
2-to-10	B(BR20AX)	Latching	10	-	120

*GE = General Electric F = Filtors, Inc. PB = Potter & Brumfield B = Babcock

COMMANDS (Cont'd)

Voltage and Impedance Characteristics

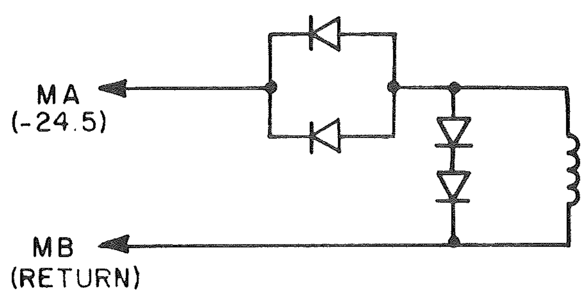
The voltage and impedance characteristics of the A and B Command Matrix drivers are:

Driver	Open Circuit Output (volts)		Source Impedance (ohms)	Maximum
	Energized	Non-energized	Energized	Load Capability (milliamps)
A	-23.5 ± 1.0	- 1.0 ± 1.0	30 ± 5	200
B	- 0.5 ± 0.5	-24.5 ± 1.0	30 ± 5	200

Command Distribution Restraints

- BACKUP AUTOMATIC FUNCTIONS WITH STORED COMMANDS - Stored command must be used for backing up critical automatic functions (transmitter turnoffs, calibration sequencing, etc)

- DIODES - Use steering and suppression diodes as shown.



NOTE

Subcontractors must use the indicated designations (MA for -24.5 VDC and MB for the return) on all hardware.

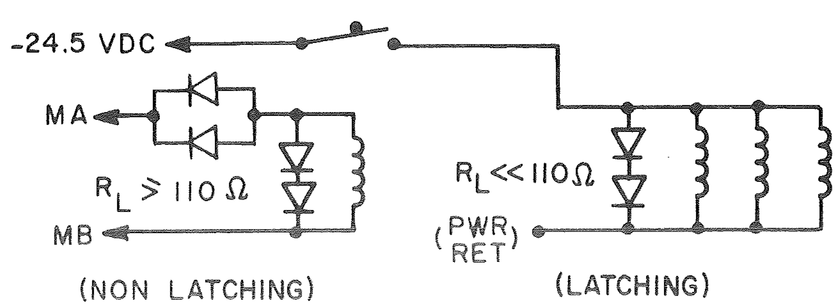
- INPUT LEADS - All input leads (MA and MB lines) to command relays must be wired to separate pins.

- NO GROUNDING - Input leads (MA and MB lines) must NOT be grounded (both MA and MB lines are pulsed to actuate command relay).

- SEPARATE CONNECTORS - All command input lines (MA and MB lines) must be wired to a connector separate from connectors used for other functions.

- OVERLOADING

  Do not overload the command matrix by using loads less than 110 ohms (parallel combinations of relay coils). Where necessary, use a relay driver as shown below. (It is obvious that two relays can be connected in parallel without requiring use of a relay driver if relays with high coil resistances are used.)



- INTERNAL COMPONENT CROSS-STRAPPING

  Cross-strapping of MA or MB lines within modules is desirable to minimize complexity in the spacecraft harness and to reduce the number of pin connections on the experiment. The experimenter shall submit a request for the total number of required commands. The NIMBUS program will provide a wiring diagram for cross-straping

Section IV

DATA HANDLING

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DATA HANDLING SYSTEM

Description

The data handling system consists of the following subsystems:

- VIP

  An internally redundant Versatile Information Processor (VIP) multiplexes low data rate experiment data and spacecraft and experiment telemetry. VIP data is transmitted in real time over one of the two VHF beacon transmitters and is stored on one of the HDRSS tracks.

- HDRSS

  Two High Data Rate Storage Systems (HDRSS) are provided for data storage. Experiment and VIP data are played back and transmitted to ground over one or both of the on-board S-Band transmitters whenever the spacecraft passes over a Command and Data Acquisition (CDA) station.

Optional System

If experiment data rates exceed HDRSS capabilities, additional data handling hardware may be provided. This may entail transmitting the data of very high data rate experiments in real time only when the spacecraft is over a CDA station. Another possibility is use of a very high data rate recorder to provide mapping of selected areas of the earth's surface, as opposed to the continuous coverage provided by the HDRSS.

VIP/HDRSS Capabilities

- VIP

  The VIP is capable of handling up to 4 kbps of digital data or an equivalent analog signal in its normal mode of operation. However, because the VIP must multiplex the data outputs of several experiments as well as both experiment and spacecraft telemetry, the data rate of any one experiment handled by the VIP is typically much lower than 4 kbps.

- HDRSS

  The HDRSS is capable of handling up to 4 kbps of digital data and analog data in the 0 to 1800 Hz range. Each HDRSS includes a 5-track tape recorder.

- TELEMETRY

  Telemetry (housekeeping or engineering data) is typically handled by the VIP, while sensor data (scientific or experiment data) may be handled by the VIP, the HDRSS, or the optional system. In addition to sensor data, the HDRSS or optional system data must include the time code and bench marks (telemetry data essential to the interpretation of sensor data). More specific rules are covered in the remaining parts of this section.

- SYNCHRONIZATION

  All experiments scanning (or cycling) must be synchronous with the VIP major frame rate of once every 16 seconds or a multiple thereof (2 scans/16 seconds, 3 scans/16 seconds, etc.).

TELEMETRY/VIP SYSTEM

VIP Characteristics

The Versatile Information Processor (VIP) System monitors telemetry and the data output of low data rate experiments (see VIP functional diagram). VIP data is transmitted real time over the beacon and stored by the HDRSS for playback when the spacecraft passes over a Command and Data Acquisition (CDA) Station. Output data rate of the VIP is 4000 bps.

Programming

Instead of a hardwired approach, the VIP utilizes a memory and associated logic as the controlling unit for generating required sampling sequences. Up to four different programs of sampling format can be used for different modes of operation or different phases of the mission, and one of these programs can be altered by ground programming.

Telemetry Accepted

The VIP system accepts analog, digital "B," and digital "A" telemetry with a variety of available sampling rates as long as the telemetry is in the required format.

- ANALOG TELEMETRY

  Analog telemetry must be such that the 0 to -6.375 vdc range corresponds to the maximum excursion of the measured parameter. The VIP converts the analog signals to a 10- bit word, thus providing 6.25 mv resolution. External noise sources limit the accuracy to 8 bits. Critical experiments should use digital "A". Analog telemetry is typically used for both housekeeping and experiment data.

- DIGITAL "B" TELEMETRY

  "B" telemetry consists of one bit words. The "off" condition must be -0.5 ± 0.5 vdc. The "on" condition must be -7.5 ± 2.5 vdc. Digital "B" telemetry is typically used only for housekeeping data.

- DIGITAL "A" TELEMETRY

  "A" telemetry consists of 10-bit words, which are read into the VIP system serially. Use of digital "A" telemetry requires approval by the spacecraft manager. Digital "A" telemetry is typically used for experiment data, with exceptions given under "Data Format." Digital "A" provides 10-bit accuracy at the experiment.

Noise

The noise on any telemetry output (analog, digital "A", or digital "B") must not exceed 5 millivolts peak-to-peak.

Capacity

The formatting unit will provide for the multiplexing and sampling of 976 distinct input signal channels from experimental and housekeeping sources, apportioned as follows:

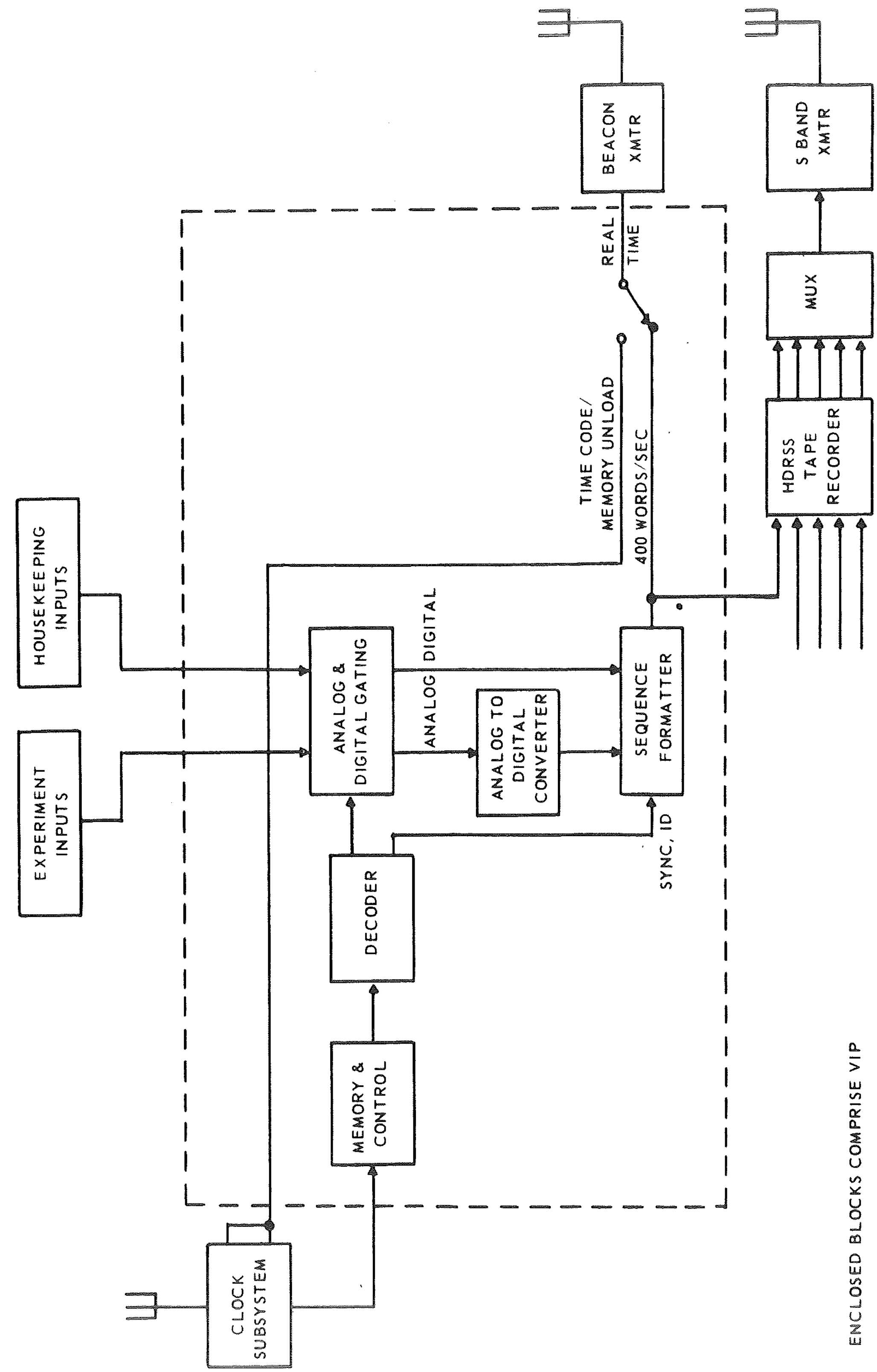
No. Inputs	Type
640	Analog
16	Digital "A"
320	Digital "B"
976	

Requests

Requests for VIP allocations must be submitted to the GSFC spacecraft manager and the spacecraft contractor and updated as new requirements are ascertained. The following information must be submitted with the request:

- TYPE OF TELEMETRY, (Analog, Digital "A," or Digital "B")
- SAMPLING RATE
- WHEN MONITORED (When must the telemetry be monitored, and when is it desired?)

Versatile Information Processor



TELEMETRY/VIP SYSTEM(Cont'd)

Data Format and Synchronization

Data Matrix (Major and Minor Frames)

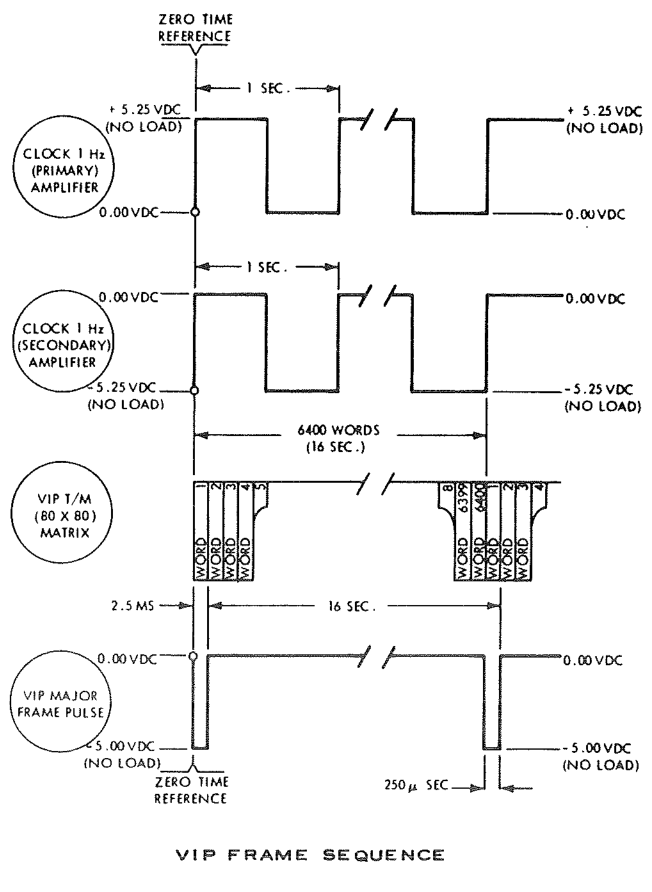
- MAJOR FRAME interval is that period of time in which every input to be sequenced in a particular program has been sampled at least one time. Major frames occur at a rate of one every 16 seconds.
- MINOR FRAMES are composed of 80 ten-bit data words, and occur at a rate of one every 0.2 seconds. Each major frame contains 80 minor frames.
- DATA MATRIX - The major frame can be regarded as a matrix of 10 bit data words with minor frames forming the rows of the matrix.

Major Frame Pulses

Major frame pulses are available (upon request) to enable experimenters to synchronize their systems to the VIP sampling sequence, with the negative going edge of the pulse coincident with the start of each major frame. Characteristics of major frame pulses are:

- Duration: 250 microseconds
- Amplitude: -5 ± 0.8 volts during pulse (0 ± 0.8 volts otherwise)
- Coherency The major frame pulse is coherent with the 1hz clock signal as indicated below
- Output Impedance: 1000 ohms or less
- Rise and Fall Times: Less than 1 microsecond

The relative timing of major frame pulses, clock signals, and VIP data words is shown in the accompanying sketch. Requests for specific time slots relative to the major frame pulse should be made as early as possible, and require approval by the GSFC spacecraft manager. A separate buffered output is provided to each user of a major frame pulse.



VIP FRAME SEQUENCE

Synchronization

- EXPERIMENT SCANNING (CYCLING)

  Experiment scanning (cycling) must be synchronous with the VIP major frame rate of one major frame every 16 seconds, or a multiple thereof (2 scans/16 seconds, 3 scans/16 seconds, etc.). This applies for all experiments, including those whose data is handled by the HDRSS. Major frame pulses and clock signals are available to facilitate synchronization. (This use requires NASA GSFC spacecraft manager approval).

- ANY OTHER DATA CYCLING RATE requires NASA GSFC spacecraft manager approval.

- SLOWER SYNCHRONOUS RATE

  In the event that one of the above rates cannot be attained, a slower synchronous rate (1 scan/32 seconds, 1 scan/48 seconds, etc.) is preferred.

- DIGITAL "A" TELEMETRY

  Digital "A" telemetry must include at least one monitor per major frame to indicate the relationship between the data and the scanning performed if the data cycling rate is not a preferred rate (n cycles/16 seconds where n = 1, 2, 3, etc.).

Calibrations

Calibrations must occur coincident with the start of a major frame or some fixed lag from that time, even if calibrations occur less frequently than once every 16 seconds.

Failure of Major Frame Pulses

If a major frame pulse fails to occur, experiments must continue to put out data. A possibility is to use clock pulses in a backup mode. Sensor data is not required to be synchronous in the event of such a failure (although it is desirable).

Bench Marks

Bench marks must be included in the grouping of VIP major frame columns comprising the sensor data allocations in the case where sensor data is handled by the VIP (Digital "A" or Analog). Bench marks are telemetry data which are essential to the interpretation of sensor data (the start and end of calibration, the start and end of spatial or spectral scanning, etc.). Otherwise, all VIP data must be processed to interpret the sensor data, which would necessitate a considerable increase in the computational facilities over those presently available at the integration and test and flight operations facilities.

Functional Data

All functional (telemetry) data must be included in the grouping of VIP major frame columns comprising the VIP Digital "B" and Analog allocations to facilitate status checks during integration and test and on-line analysis of experiment performance during flight operations. Bench mark data must therefore be handled by both VIP Digital "B" and VIP Digital "A" (or Analog) or HDRSS, whichever is used for the sensor data. On-line analysis (at NTCC) is required so that any corrective action required can be taken while the spacecraft is within the coverage of a CDA station. Adherence to this rule is necessitated by the difficulties of stripping selected data from the VIP data stream, the limited bandwidth of the Ulaska-NTCC data link, and the limited data processing facilities available at NTCC and the spacecraft contractor site. A possible exception to this rule is refined functional data such as linearity checks which need only be included in the sensor data output (provided the Digital "B" or Analog telemetry data is sufficient to indicate experiment status without this check). Any other exception requires NASA GSFC spacecraft manager approval.

TELEMETRY/VIP SYSTEM (Cont'd)

Impedance Requirements and Other Restraints

Analog and Digital "B" Impedance Requirements

ANALOG - Experiment or telemetry output impedance shall not exceed 10 K ohms. The input impedance presented by the VIP to each analog signal will be 1 megohm during sampling and 10 megohms during nonsampling or turn-off time.

DIGITAL "B" - Experiment or telemetry output impedance shall be 1 megohm or less in the ON condition and 50K or less in the OFF condition.

Other Restraints

- UNDER NORMAL OPERATING CONDITIONS, the Digital "B" and Analog inputs to the VIP shall not exceed the -10 to +0.5 vdc range.
- UNDER FAULT CONDITIONS, the Digital "B" and Analog inputs to the VIP shall not exceed the -25 to +0.8 vdc range. If opening the telemetry output resistor would develop a large voltage to the multicoder input, parallel the output resistor with protective (Zener diode) circuits to keep the output below -24 volts (more positive than -24 volts) for all conditions of failure modes.
- LEAKAGE CURRENT from the VIP to any analog input may be as high as 1 μa during sampling time and 50 na during nonsampling time. The experiment must be capable of operating without degradation under these conditions, and the resultant degradation of the analog telemetry signal must be negligible.
- PROVIDE AN APPROPRIATE BLEEDOFF PATH (RESISTIVE) for capacitor terminated outputs.
- PROVIDE AN APPROPRIATE BLEEDOFF PATH for telemetry encoder in all telemetry circuits using an isolation diode in the telemetry output line.
- PROVIDE ISOLATION from subsystem operating circuits in case of telemetry shorting. Document the effects of an open and a short on each telemetry output for use by the GSFC spacecraft manager and the spacecraft contractor.
- PROVIDE LEADS for connection to external telemetry power and ground if the telemetry function can be isolated from its own internal power bus. For example, a thermistor driven by a subsystem -24 volts through a dropping resistor to subsystem ground can be driven by -24 volts from a telemetry supply through telemetry ground.
- SEPARATE CONNECTORS are to be provided for Analog signals, Digital "B" signals, and major frame pulses, except that major frame pulses can share a connector with A_1 timing pulses for Digital "A" telemetry (defined later in this section).
- ONE CONNECTOR PIN is to be used for each Analog and Digital "B" telemetry point. Return is through power supply ground unless the ground for a particular transducer is isolated from power supply ground, in which case a connector pin is to be wired to each distinct signal ground point.
- ONLY ONE WIRE is provided for each experiment utilizing major frame pulses, and thus only one connector pin should be wired for major frame pulses.

VIP Gate Failure

Experiments must be capable of operating without degradation in the event of a VIP gate failure. If this occurs, the voltage fed back to an experiment on the VIP to experiment lead may be as low as -10 volts or as high as +6 volts full time and may be as low as -24 volts for approximately 200 microseconds during sampling.

VIP Digital "A"

10-Bit Serial Words

VIP Digital "A" input gating enables the transfer of a maximum of 10 bits of digital data in a serial fashion to the VIP. If 8-bit accuracy is acceptable, analog telemetry should be used since the VIP Digital "A" capacity is limited to 16 distinct input signals.

Use

VIP Digital "A" has been found most useful for monitoring the output of low data rate experiments and for monitoring the calibration mode data of some experiments.

A/D Converters

If an analog-to-digital converter is used so that Digital "A" can be used in conjunction with an analog monitor, a calibration mode must be provided to check the A/D converter. Digital "A" and Digital "B" indications of when calibration is being performed must also be given.

Timing Pulses

Because VIP Digital "A" data must be read into the VIP serially with fixed timing restrictions, the VIP system provides to the experiment the following pulses which the experiment must use to synchronize its data with the VIP.

Pulse	No. Pulses	Rise Time (microsec)	Duration (microsec)
A_1	1	6	2500
B_1	1	1	50
C_1	10	1	10

These pulses have a positive peak value of 5.0 ± 0.8 volts.

Data Pulses

The 10 data (D_1) pulses provided by the user must have a maximum rise time of 1.0 microsecond, a duration of 10 microseconds and a period of 100 microseconds. For the data pulses:

- The "zero" state must be 0 ± 0.8 volts
- The "one" state must be 5 ± 0.8 volts.

Under Fault Conditions

Under fault conditions, Digital "A" inputs to the VIP shall not exceed -1.0 or +8.0 vdc.

Connectors

- B_1, C_1, and D_1 PULSES

  Leads for B_1, C_1, and D_1 pulses must be wired to separate Twin-Ax connectors. Experiments requiring more than one Digital "A" monitor must provide separate circuitry and connectors for each function monitored.

- A_1 PULSES

  Cannon D connectors must be used for A_1 pulses, with separate pins for each function monitored. Major frame pulses may use the same connector as A_1 pulses.

TELEMETRY /VIP SYSTEM(Cont'd)

VIP Digital "A" (Cont'd)

Synchronization

Synchronization of Digital "A" data transmission must be achieved in the manner specified diagrammatically (see sketch below).

- DIGITAL "A" DATA

  All Digital "A" data must be transferred to the VIP during the Digital "A" pulse.

- MOST SIGNIFICANT BIT

  The user's first (most significant) bit of information must reach 90% of its value a minimum of 90 microseconds before the leading (positive going) edge of the first C_1 pulse.

- DURATION OF MSB

  The most significant bit shall remain at its level at least until the leading edge of the first C_1 shift pulse signal occurs.

- 2nd TO 10th BITS

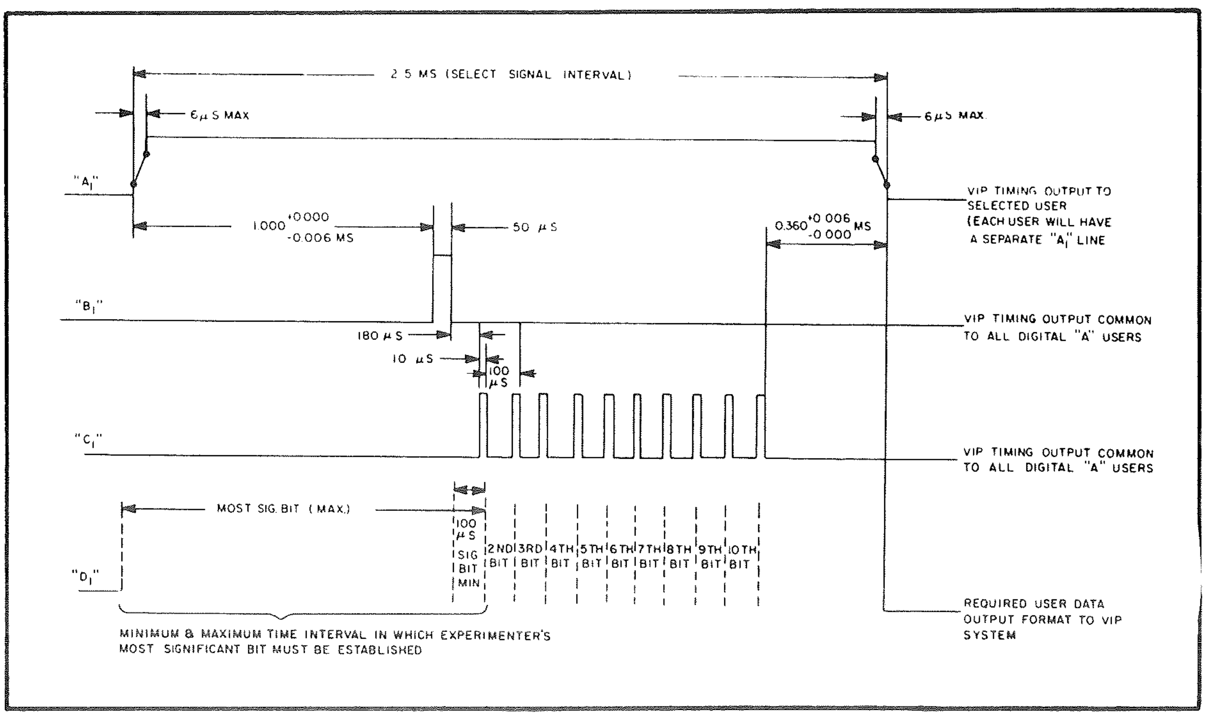
  The second through tenth most significant bits shall reach 90% of value less than 20 microseconds from the time the 10% point on the falling edge of the previous C_1 pulse is reached.

- DURATION

  These bits must remain at their levels at least until the leading edge of the next C_1 pulse is reached.

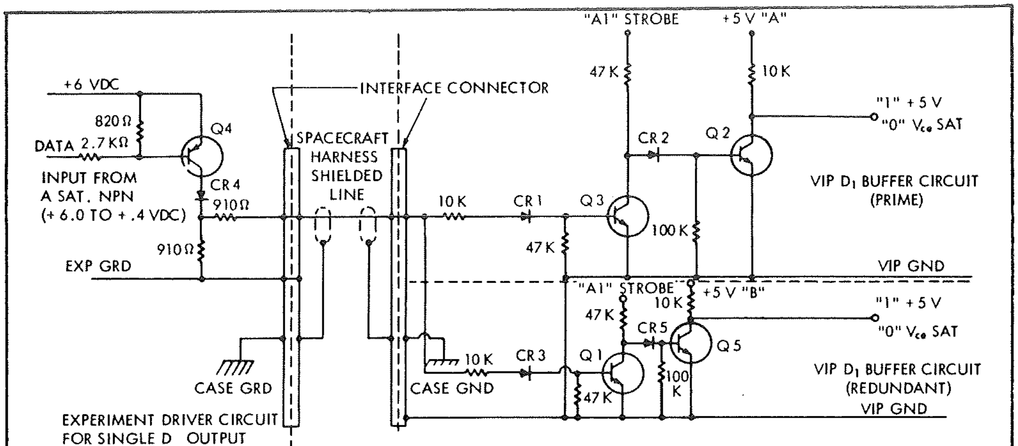
- WHEN NOT SAMPLING

  During all times the experiment is not being sampled (i.e., when the A_1 pulse is at 0 ± 0.8 volts), the user's information output signal (D_1 pulses) shall be 0 ± 0.8 volts.



Experiment Driver

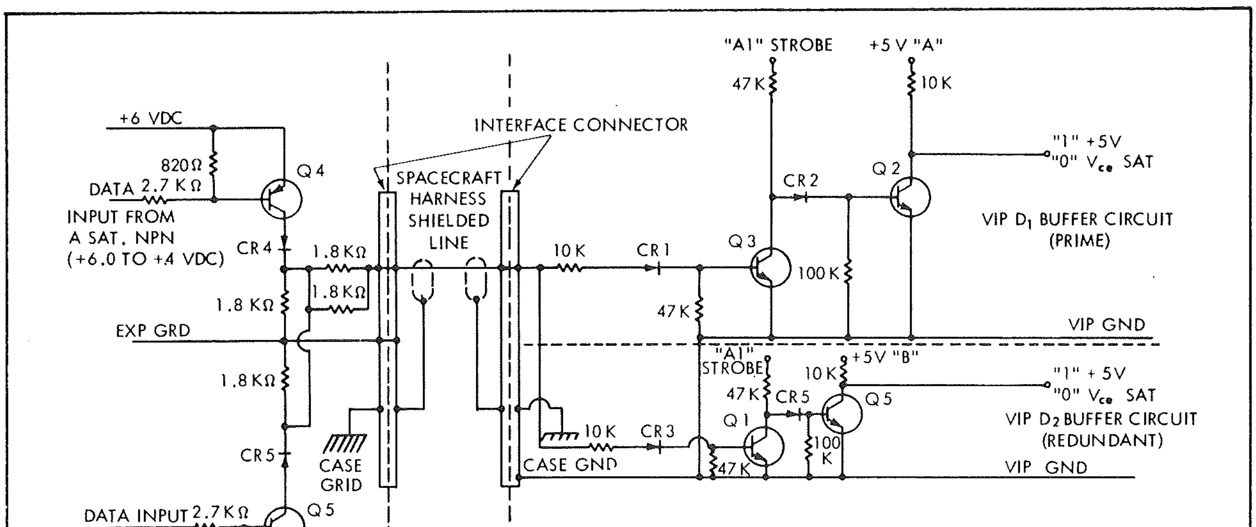
One of the drivers (electrical schematics are shown below) must be used for the transmission by the experiment of data (D_1) pulses. Use of the redundant circuit is preferred.



COMPONENT TABLE

IDENT	VALUE	TOL	QTY	TYPE	REF. DESIG.	IDENT.	STOCK NO.	ALT. STOCK NO.
RESISTOR		± 5%		0.25 W	Q 1	TRANSISTOR	2 N 2222 A	
	820		1		Q 2	TRANSISTOR	2 N 2222 A	
	910		2		Q 3	TRANSISTOR	2 N 2222 A	
					Q 4	TRANSISTOR	2 N 2907 A	
					Q 5	TRANSISTOR	2 N 2222 A	
					CR 1	DIODE	1 N 916	1 N 914
	2.7 K		1		CR 2	DIODE		
	10 K		4		CR 3	DIODE		
	47 K		4		CR 4	DIODE		
RESISTOR	100 K	± 5%	2	0.25 W	CR 5	DIODE	1 N 916	1 N 914

EXPERIMENT INTERFACE DRIVER FOR SIMPLE DIGITAL "A" D_1 CIRCUIT



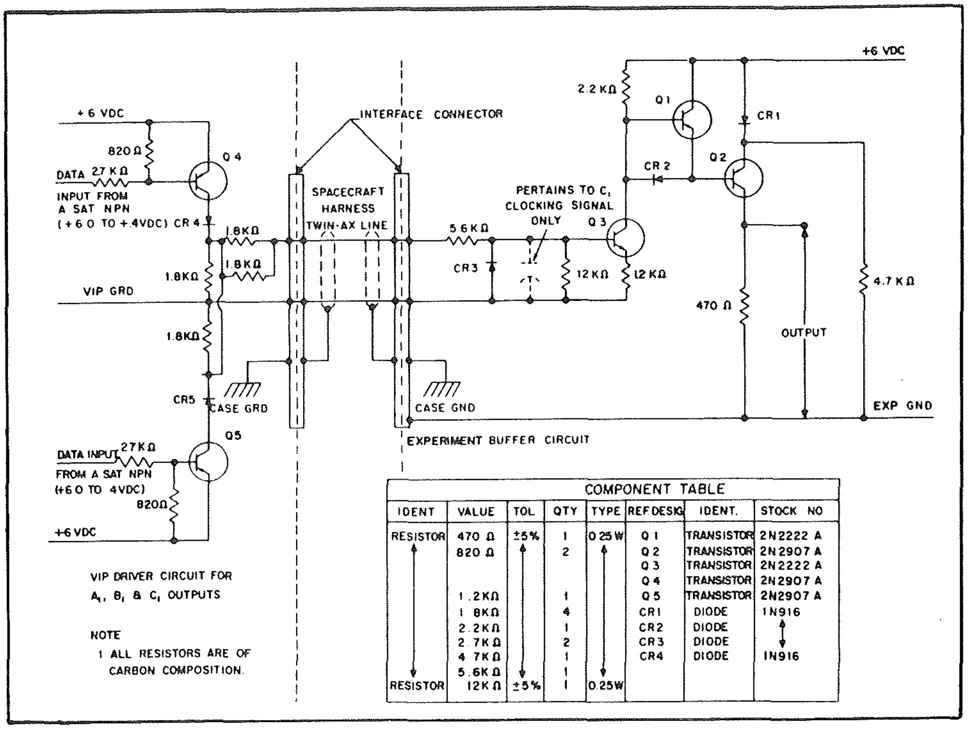
COMPONENT TABLE

IDENT	VALUE	TOL	QTY	TYPE	REF. DESIG.	IDENT	STOCK NO.	ALT. STOCK NO.
RESISTOR		± 5%		0.25 W	Q 1	TRANSISTOR	2 N 2222 A	
	820		2		Q 2	TRANSISTOR	2 N 2222 A	
					Q 3	TRANSISTOR	2 N 2222 A	
					Q 4	TRANSISTOR	2 N 2222 A	
	1.8 K		4		Q 5	TRANSISTOR	2 N 2222 A	
					CR 1	DIODE	1 N 916	1 N 914
	2.7 K		2		CR 2	DIODE		
	10 K		4		CR 3	DIODE		
	47 K		4		CR 4	DIODE		
RESISTOR	100 K	± 5%	2	0.25 W	CR 5	DIODE	1 N 916	1 N 914

EXPERIMENT INTERFACE DRIVER FOR REDUNDANT DIGITAL "A" D_1 CIRCUIT

Buffer Amplifier

The mandatory load for A_1, B_1, and C_1 pulses is the buffer amplifier whose electrical schematic is shown in the sketch below.



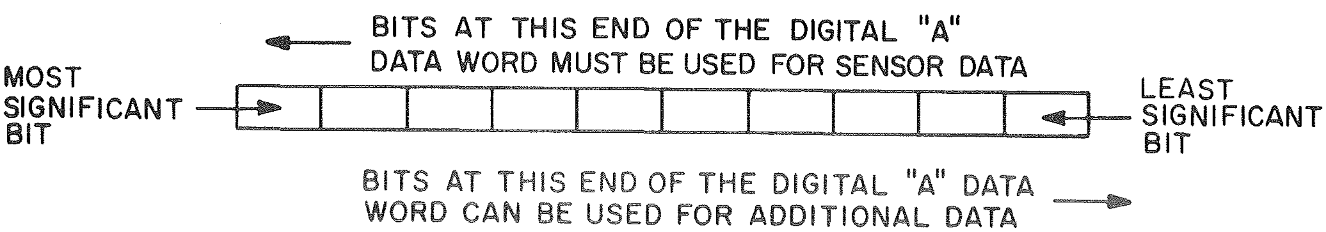
COMPONENT TABLE

IDENT	VALUE	TOL	QTY	TYPE	REF DESIG	IDENT.	STOCK NO
RESISTOR	470 Ω	±5%	1	0.25W	Q1	TRANSISTOR	2N2222 A
	820 Ω		2		Q2	TRANSISTOR	2N2907 A
					Q3	TRANSISTOR	2N2222 A
					Q4	TRANSISTOR	2N2907 A
	1.2KΩ		1		Q5	TRANSISTOR	2N2907 A
	1.8KΩ		4		CR1	DIODE	1N916
	2.2KΩ		1		CR2	DIODE	
	2.7KΩ		2		CR3	DIODE	
	4.7KΩ		1		CR4	DIODE	1N916
	5.6KΩ		1				
RESISTOR	12KΩ	±5%	1	0.25W			

EXPERIMENT INTERFACE BUFFER FOR DIGITAL A DATA, A_1, B_1 AND C_1 OUTPUTS

Use of Extra Digital "A" Bits

It is possible that Digital "A" data may be used by experiments (perhaps to simplify the data format) even in the case where the sensor resolution is less than ten bits. In this case, it is permissable to use the bits not used for sensor data for other data, provided that the pulses used for additional data are those which would normally be used as the least significant bits of the Digital "A" data word (as shown below). This rule is necessitated by the characteristics of the D/A converters used in integration and test and flight operations, which convert the eight most significant bits of the Digital "A" word into an analog trace.



TELEMETRY MONITOR SELECTION and USE

The selection and use of telemetry is one of the most critical parts of spacecraft and experiment design, not only when the spacecraft is in orbit but also during integration and testing.

- IN ORBIT

  After the spacecraft is in orbit, telemetry is the only means available for determining the state of an experiment

- DURING INTEGRATION AND TEST

  As indicated by the following listing, telemetry is the only method of determining the state of an experiment after the bench check phase of testing (because access to the experiment becomes more and more restricted). (In general, if a parameter is not available through telemetry, it cannot be ascertained by other means.)

Use of Telemetry	Measurement Objective	Analysis Tools Used
Evaluation of Functional Operation	To verify normal operation and mode switching	Computer Programs, AGE Installations, Status and Command Verification, Temperature Set, Brush Decommutators, Matrix Display, Data Listing, Limit Comparison (limit checks all spacecraft functions), and Analysis Programs for each Subsystem
Experiment Calibration	To perform thermal vacuum calibration with calibration targets	Subsystem Analysis Programs, Temperature Set, Plotting Programs (meters, oscilloscopes, and photo processors)
Interference Analysis	To verify subsystem electrical, mechanical, and RF compatibility	Subsystem Analysis Programs, Brush Decommutators, Plotting Programs, Limit Comparisons, and Data Listings
Performance Stability	To determine long term subsystem functional repeatability	General Averaging Programs in conjunction with Plotters, Subsystem Analysis Programs, History Programs, and an Operating Time Summary
Malfunction Analysis	Fault isolation and identification	All Computer Programs

Experiment Protection

Telemetry protects the experiment not only by facilitating the detection of anomalies within the experiment but also by determining if other experiments or the spacecraft degrade experiment performance. The list above shows the measurement objectives of telemetry and the analysis tools which the spacecraft system contractor has developed to facilitate the detection of anomalies and incompatibilities. The following measurement requirements must therefore be satisfied (as a minimum) by the selected monitors:

- EVENT VERIFICATION AND STATUS DETERMINATION
- FAULT IDENTIFICATION AND ISOLATION
- CALIBRATION
- FUNCTIONAL SIGNATURES (what is expected)
- LONG TERM PERFORMANCE STABILITY

Spacecraft contractor personnel are available to assist in determining if the telemetry monitors selected meet these goals.

Their involvement will help to attain maximum protection of the experiment through the following:

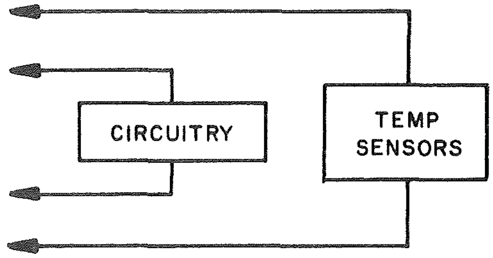
- IMPROVEMENT OF TELEMETRY MONITOR SELECTION through the experience gained from four NIMBUS Spacecraft.
- BETTER DEVELOPMENT OF THE ANALYSIS TOOLS for the particular needs of each experiment. Only the information gained by a continuing interface can make this possible.

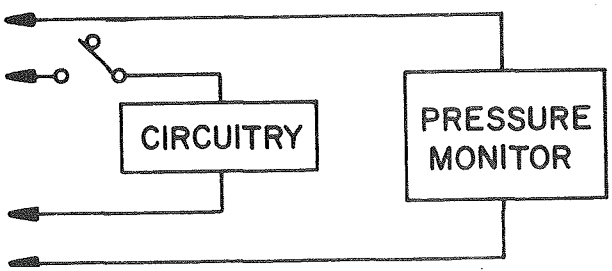
Specific Rules

Experience gained from four NIMBUS Spacecraft has shown that the mandatory rules listed on the remaining pages of this section help to ensure that the above requirements are met. Selection and use of telemetry monitors in accordance with the following rules, and the above requirements, is the only way that adequate protection of the spacecraft and experiments can be gained.

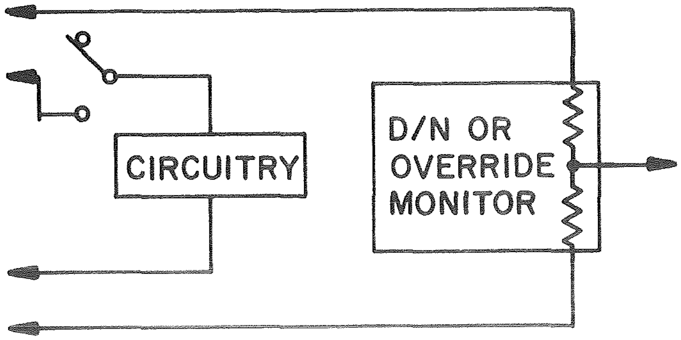
TELEMETRY MONITOR SELECTION and USE (Cont'd)

- PROVIDE MOTOR DRIVE CURRENT MONITORS in place of voltage monitors.
- DO NOT PROVIDE SUBSYSTEM INPUT CURRENT MONITORS OR FUSES. These items will be included in the spacecraft electrical system by the spacecraft contractor (for accessibility to fuses and to minimize the number of current monitor calibration curves).
- COMMAND RELAYS - Provide Digital "B" relay state monitors on all command relays.
- PROVIDE LEADS for connection to external telemetry power and ground if the telemetry function can be isolated from the experiment power bus. Example: thermistor circuits (See sketch).
- DC/DC CONVERTERS - Provide analog monitors for DC/DC converter output voltage.
- PROVIDE TEMPERATURE MONITORS WHERE:
  - Components are susceptible to overheat damage
  - Components experience significant temperature transients during operation
  - Data output is temperature dependent
  - Provide at least one temperature monitor in each component
  - Provide hot and cold temperature monitors in each module
  - Provide temperature monitors in association with all pressure monitors

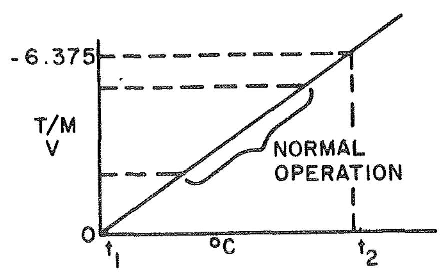




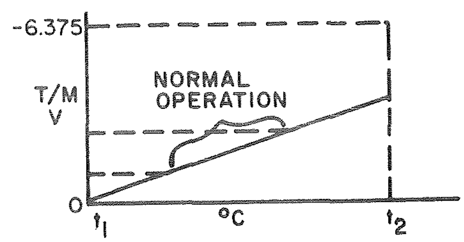
- TWO STATE EVENTS (ON/OFF) - The "ON" state or the occurrence of the event being monitored must be denoted by a logical "one," and the "OFF" state or absence of the event being monitored must be indicated by a logical "zero."
- MAKE DAY/NIGHT AND DAY/NIGHT OVERRIDE MONITORS independent of subsystem power (see sketch).



- PRESSURE MONITORS - Provide pressure monitors on all pressurized units. Power to these monitors must be full time (See sketch).
- KEY SIGNALS - Provide monitors of the presence of key signals supplied by the clock.
- PROVIDE MONITORS of key points in formatting and sequencing logic circuits.
- PRIME DATA OUTPUTS - Monitor prime experiment data outputs prior to processing.
- CALIBRATION CYCLES - Provide Digital "B" telemetry to indicate start and end of calibration cycles.
- AUTOMATIC MODE SWITCHING - Provide Digital "B" monitors for relay and solenoid movements associated with automatic mode switching.
- SCALE THE RANGE of the analog monitors to make maximum utilization of the 0 to -6.375 volt telemetry range (see sketch).

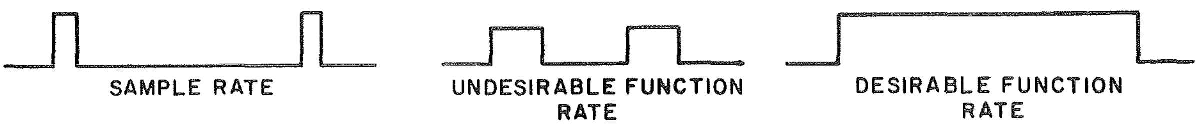


INSTEAD OF



- TELEMETRY MONITOR CALIBRATION CURVES - Establish calibration curves for each telemetered function. Calibration curves must be furnished with each experiment model supplied to the spacecraft contractor (including preprototypes).

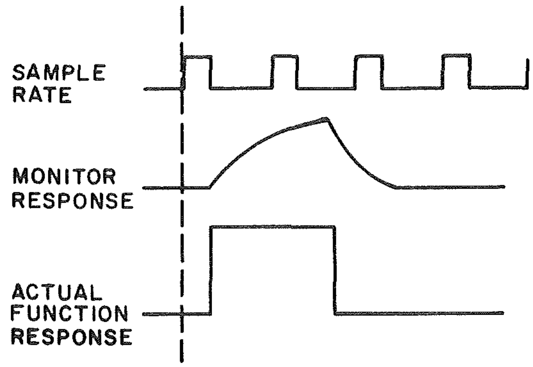
- SELECT FOR TELEMETERING the functions that have a frequency of occurrence and duration compatible with the available VIP sampling rates. See example below.



- USE PULSE STRETCHER TYPE CIRCUITRY to extend the time of short duration critical functions that must be telemetered.

- TEMPERATURE SENSITIVE MONITORS - Provide monitors which are not temperature sensitive. Temperature compensation must be provided if the monitoring element is temperature sensitive.

- RESPONSE TIME OF MONITORS - Design the response time of the monitor to be compatible with the sampling rate used and the function to be monitored. Do not employ monitors with the response time/sample rate shown at right.



- SAMPLE RATE. Be careful when defining the sample rate, especially for fast changing signals. Sufficiently high sampling rates must be used in order to prevent "aliasing", which is a distortion of the reconstituted signal which occurs when the energy in the sampled signal at frequencies in excess of one-half the sampling frequency is not negligible. In order to determine the sampling rate, it is imperative that the expected frequency spectrum of the sampled signal be determined first, rather than basing the sampling rate on an arbitrary decision. (Reference: H.L. Stiltz, Aerospace Telemetry, Vol. I, Prentice Hall, 1961, pp. 86-91).

- RETURN TO ZERO VOLTS. Design analog monitors to go to approximately 0 volts when function being telemetered is de-energized. For example: Design subsystem power supply voltage monitors to go to 0 volts when the power supply is OFF. The telemetry monitor must, of course, be calibrated in accordance with this rule.

- SAME SOURCE. Energize a telemetry monitor from the same source as the function being monitored. Do not energize a telemetry monitor from a power source which may be turned off while the function being monitored is still energized. (Example--monitoring the rotation of a mirror with power supplied from the same source as an electronics package which could be turned off although the mirror is kept rotating).

- OPERATING CIRCUITS. Provide isolation from subsystem operating circuits in case of telemetry shorting. Document for use by the spacecraft manager and spacecraft contractor the effects of an open and a short on the telemetry output.

- MOTOR DRIVEN COMPONENTS. Provide Digital "B" telemetry for monitoring the rotation of mirrors, gears, and other motor driven components.

- STATUS AND MODE INDICATORS. Use Digital "B" monitors for status and mode indicators.

- UNAMBIGUOUS DATA. Monitor functions which provide unambiguous data or monitor additional functions which allow resolution of ambiguities when considered together.

- FAILURE ANALYSIS. Select the telemetry monitors needed as data for failure analysis. Check with the spacecraft system contractor so that past experience can be used to ensure that the monitors used will be adequate for failure analysis.

- TELEMETRY FUNCTIONS. Backup telemetry functions which are needed for data analysis with a redundant monitor. Check with the spacecraft system contractor to see where redundant monitors are essential.

- ANALOG TO DIGITAL CONVERTERS. Provide an in-flight calibration check of analog-to-digital converters.

- DIGITAL "A" HOUSEKEEPING DATA. Provide backup housekeeping data for any housekeeping data transmitted on Digital "A".

- GROUND POTENTIAL SENSITIVITY. Telemetry circuits must be designed and/or grounded so that changes in ground potential level have little or no effect on the telemetry monitor output levels.

- CLEARING OF TELEMETRY REGISTERS. Clearing of registers holding functional monitor data must be performed by the actuating source, such as command relays or internal circuitry, so that the register indicates the actual status of the telemetry point being monitored. Clearing should not be performed by clock pulses or major frame pulses when they bear no relationship to the status of the function being monitored. Example: A register indicating an experiment is in a calibration mode is cleared by a major frame pulse but the experiment remains in calibration until acted upon by a command relay. The command relay should have cleared the register, rather than the major frame pulse.

- USE DIGITAL "B" FOR RELAYS. Do not use an analog monitor to denote a relay state or multiple relay states (via a resistor bank, as is often done). Not only does this tie up the limited VIP analog capability, but only four levels of analog telemetry are recognized in some integration and test and flight operations activities, and relay data gained in this manner can be inconclusive.

- RANGE OF OPERATION. Only one range of operation (and thus one calibration curve) is to be used for each analog or Digital "A" telemetry slot. Exception to this rule requires spacecraft manager approval. If approval is granted, the range of operation must be indicated by both analog or Digital "A" and Digital "B" telemetry.

HIGH DATA RATE STORAGE SUBSYSTEM (HDRSS)

HDRSS Characteristics

- 2 UNITS

  Two identical HDRSS units (including tape recorders) are provided on the spacecraft for storage and playback of experiment data. These may be used in a redundant manner, or each may store data from different experiments.

- 5 CHANNELS

  Five channels are provided on each HDRSS; one reserved for time code and one reserved for VIP. Thus two analog and one digital channel are available for experiment data.

- STORAGE CAPABILITY

  The storage capability of each HDRSS, is nominally 134 minutes (approximately 1.5 orbits for a 600 nm circular orbit).

- STORED DATA

  Data stored over either HDRSS is played back and transmitted over one of the redundant S-Band systems (see sketch next page) whenever the spacecraft is within the coverage cone of a command and data acquisition (CDA) station.

- RECORD/PLAYBACK RATIO

  Record/playback ratio of 1/32 results in data for a full orbit. The data is stored on either HDRSS to be played back and transmitted to a Command and Data Acquisition (CDA) station in approximately 3 minutes. CDA stations are located at Gilmore Creek, Alaska and Rosman, North Carolina; the spacecraft is not acquired by a CDA station during an average of two orbits daily (blind orbits).

- RESERVED CHANNELS

  Channel 1 on both HDRSS units will be reserved for storage of the VIP system data output (low data rate experiment outputs and spacecraft telemetry). A second channel is reserved for time code.

Experiment Requirements

- SYNCHRONIZATION

  Experiment scanning or cycling must be synchronous with the VIP major frame rate of once per 16 seconds or a multiple thereof (2 scans/16 seconds, 3 scans/16 seconds, etc.). VIP major frame pulses and clock signals are available to facilitate synchronization.

- 36-BIT TIME CODE

  NASA minitrack 36-bit time code (available from the clock) must be included in the sensor data output of all experiments whose sensor data is not handled by the VIP. This is required so that an additional track of each HDRSS can be used for the recording of experiment data and to reduce the data processing required at the integration and test and flight operations facilities.

- CALIBRATION DATA

  Calibration data (sensor outputs during calibration) must be included in the VIP data, as well as the HDRSS data of experiments whose sensor data is handled by the HDRSS.

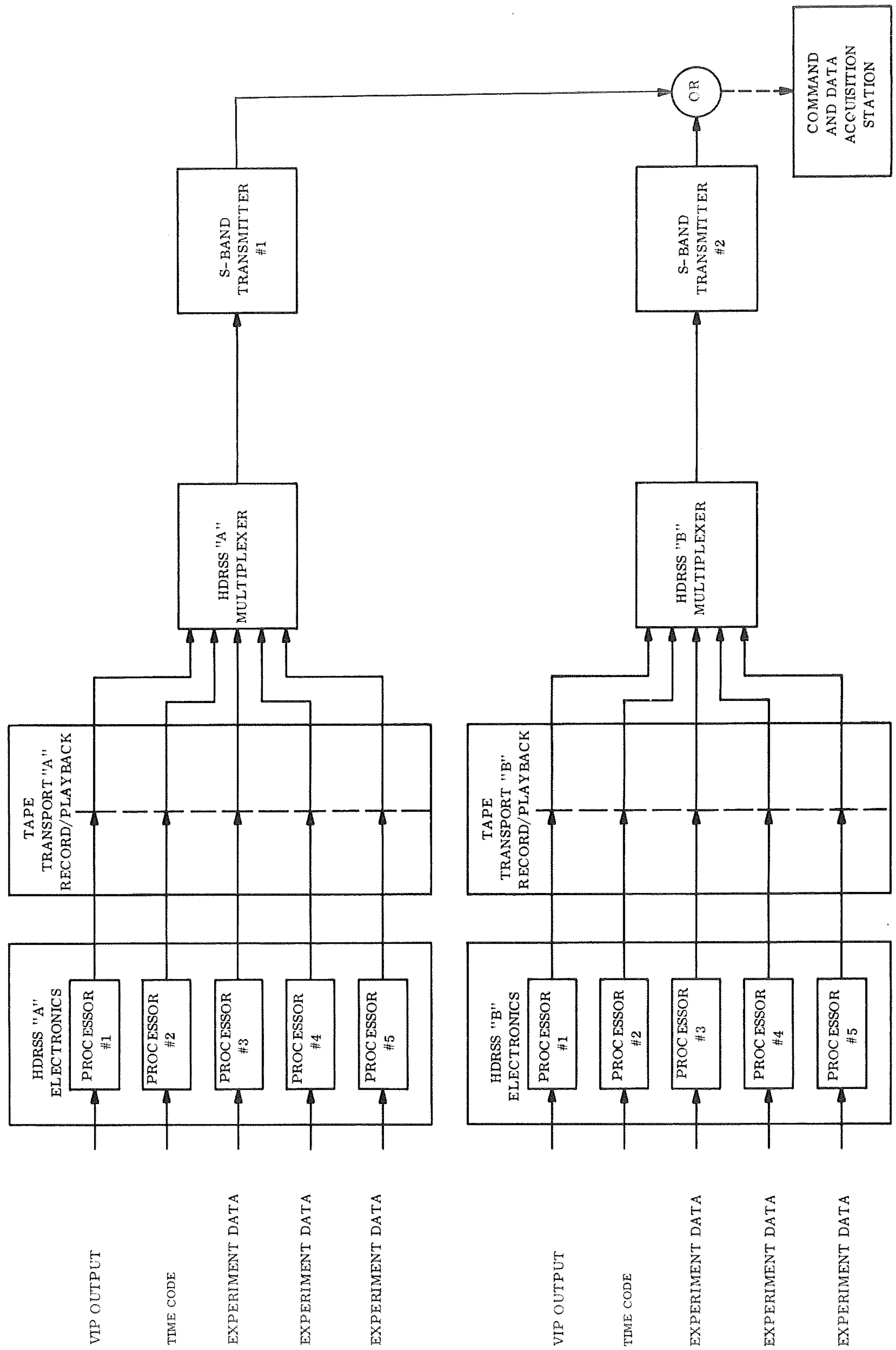
- BENCH MARKS

  Bench marks must be included in sensor data to the HDRSS. Bench marks are telemetry data which are essential to the interpretation of sensor data such as the start and end-of-calibration, the start and end of spatial or spectral scanning, etc. Otherwise, VIP data would have to be used in the interpretation of sensor data, requiring an increase in the computational facilities at the integration and test and flight operations sites.

- EXPERIMENT INPUTS TO THE HDRSS must be dual so that the two units can be used in a redundant manner and must be suitably isolated so that if a short circuit or other malfunction of the HDRSS occurs, the remaining experiment output to the other HDRSS can be used without system or performance degradation.

- TWISTED PAIR LEADS must be used for the transmission of experiment data to the HDRSS.

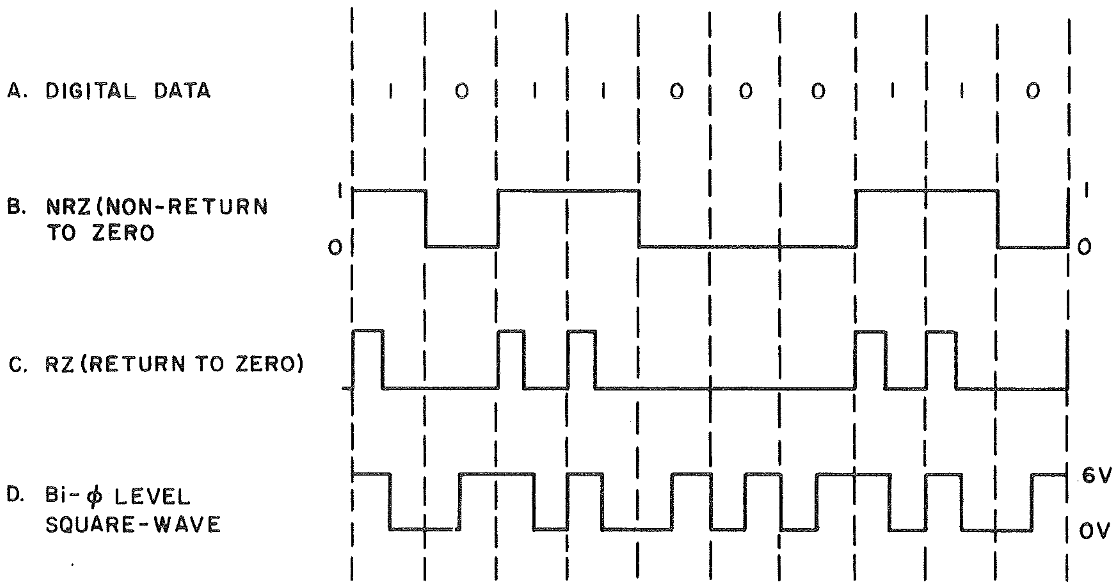
HDRSS BLOCK DIAGRAM



HIGH DATA RATE STORAGE SUBSYSTEM (HDRSS) (Cont'd)

Biphase

- DIGITAL DATA inputs to the HDRSS must be in biphase form. In the biphase-level square wave a decreasing voltage in the center of the bit represents a one; an increasing voltage represents a zero (see below).



HDRSS Channel Characteristics (Typical NIMBUS D)

Channel No.	Signal Type	Data Rate/ Frequency Response	Permissable Input Voltage Variation	HDRSS Input Impedance	Experiment Source Impedance	HDRSS Noise Specifications
1	digital biphase level	3.75 kbps	0 (±0.6) to 6.0 V/open circuit	22 ± 4 kohms, ac coupled	600 ohms (nominal)	Equivalent to 1 bit error in 10^5 bit.
2	digital biphase level	4 kbps	0 (±0.6) to 6.0 (+0.5, -1.0) V/open circuit	22 ± 4 kohms	600 ohms (max.)	Equivalent to 1 bit error in 10^5 bits
3	analog	0 to 1800 Hz	-2.3 (±0.1) to -8.0 (±0.1) V	50 kohms min. at -3.5 ± 0.5V	600 ohms (max.)	If experiment SNR is 34 db pk-pk/rms, SNR at CDA station is 30 db pk-pk/rms.
4	analog	0 to 345 Hz	0 (+0.00, -0.25) to -6.0 (±0.25) V	50 kohms min. at -3.5 ± 0.5V	600 ohms (max.)	If experiment SNR is 35 db pk-pk/rms, SNR at CDA station is 30 db, pk-pk/rms
5	amplitude modulated subcarrier	100 bps (2.5 kHz subcarrier)	1.5 (±0.2) V pk-pk "1" 0.5 (±0.1) V pk-pk "0"	500 ohms min. at 2.5 kHz	50 ohms (max.)	Compatible with system performance requirements

Section V

SPECIAL PRECAUTIONS - ELECTRICAL

Page No.

High Voltage Precautions 5-1
Radio Frequency Interference (RFI). 5-2
Grounding 5-6
Shielding. 5-8
Field Effect Transistor (FET) Precautions 5-10

HIGH VOLTAGE PRECAUTIONS

Use of voltages in excess of 250 volts requires the early approval of the GSFC spacecraft manager. Past experience indicates that for voltages in excess of 250 volts, the following precautions must be followed:

- SHARP CORNERS - Do not allow sharp corners on any conductors carrying high voltages or on grounds near exposed high voltages.
- DO NOT USE SEALED CONNECTORS - Ventilate high voltage connectors.
- DURING LAUNCH - Do not operate high voltages during launch. It will also be necessary to keep the experiment switched off for a sufficient length of time in orbit to permit proper outgassing.
- DE-AERATE ALL POTTING MATERIAL for conformal coated or solid potted boards after pouring,but before the potting compound sets.
- USE SILICON-RUBBER INSULATED WIRE
- ADEQUATELY VENT MODULES which house high voltages, or pressurize to maintain pressure high enough to eliminate corona.
- ADEQUATELY SEPARATE all exposed high voltages from ground, lower voltages, and voltages of opposite polarity.
- CORONA OR ARCING - The experiment must be designed so that if corona or arcing develops as a result of a command error or malfunction of the equipment, the associated effects will not violate any of the electrical interface specifications.
- CORONA OR ARCING DURING LAUNCH - Where potentials in excess of 100 volts which can cause corona are active during the launch phase, means for prevention such as pressurization, solid potting, or current limiting of power supplies must be provided. When pressurization is not used, modules must be vented to assure rapid outgassing.

RADIO FREQUENCY INTERFERENCE (RFI)

RFI Design Practices

Equipment must be designed and packaged so it is not adversely affected by interfering sources of energy and, conversely, must not be a source of interference which might adversely affect the operation of other equipment. Proper grounding and shielding, which are covered in the next two sections, are important in reducing RFI. In addition, the following guidelines must be followed:

- STABILITY - Circuits, especially of the feedback variety, must be designed so that stability is achieved at all frequencies.
- PARTICULAR ATTENTION must be given to wiring and component arrangement to avoid unwanted coupling causing undesirable feedback and spurious modes of operation.
- ADEQUATE FILTERING must be supplied for decoupling power supplies.
- LINEAR AMPLIFIERS, free of harmonic and cross-modulation distortion, must be used where applicable.
- SHORT LEADS - Leads must be kept as short as possible and cross-coupling between circuits minimized through use of shielding and right-angle crossing of leads.
- RADIATING COMPONENTS such as transformers and coils must be shielded.
- SPARK AND ARC SUPPRESSION must be used on all relays where current is broken.
- MINIMIZE RISE TIME - Coils containing rapidly changing currents such as relay coils must have effective surge-damping elements associated with them or minimize the rise time by other means.
- FILTERS must be used to eliminate unintended signals conducted on a line. Filters should be designed to attenuate to an acceptable level all frequencies both above and below the desired frequency range.
- CROSS-COUPLING between circuits shall be minimized by adherence to the following rules:

  SHIELD all low level signal leads (less than one volt peak amplitude) within modules.

  DO NOT BRING low level signal (less than one volt) leads out of modules (amplify first to at least one volt amplitude).

  POWER WIRING should be run separate from signal carrying wire.

  AUDIO FREQUENCY WIRING should be run separately from high frequency wiring. If separation is not possible because of space limitations, the audio frequency wires should consist of twisted pairs with an outer shield.

  ANTENNA WIRES should be separated from other antenna wires.

  TERMINATING AND SOURCE IMPEDANCES should be low to minimize electric field interference pickup.

- OTHER METHODS OF REDUCING RADIATION:

  ELECTROMAGNETIC COUPLING can be reduced by placing a high impedance to ground.

  ELECTROSTATIC COUPLING can be reduced by placing a low impedance to ground.

  ELECTROMAGNETIC FIELDS can be reduced by using a single point ground for the entire noise-producing circuit, and by orienting the components and leads so that their field induce a minimum of current in the surrounding metal.

  CHASSIS RADIATION is the result of induced currents circulating in the walls of metal chassis. This type of interference can be eliminated by:

  - ARRANGING COMPONENTS producing electromagnetic fields in a manner which will induce a minimum of current in the shield material.
  - FOR LOW FREQUENCY CIRCUITS, by using a common ground bus or load.
  - FOR HIGH FREQUENCIES, by using short duct leads.
  - USING A CONTINUOUS, UNBROKEN, UNSPLICED SHIELD for containing the offending field.

RFI Testing

Based on past experience (See Nimbus B Freq. Spectrum, next two pages), the following tests are required as a minimum for all engineering model and prototype model spacecraft hardware. A test plan which outlines the specific approach to RFI testing shall be submitted to the Nimbus Spacecraft Manager for review and approval prior to the beginning of the test program.

Radiation Tests: The unit under test shall be illuminated with the following power levels at the specified frequencies from an antenna placed approximately one (1) meter from the experiment.

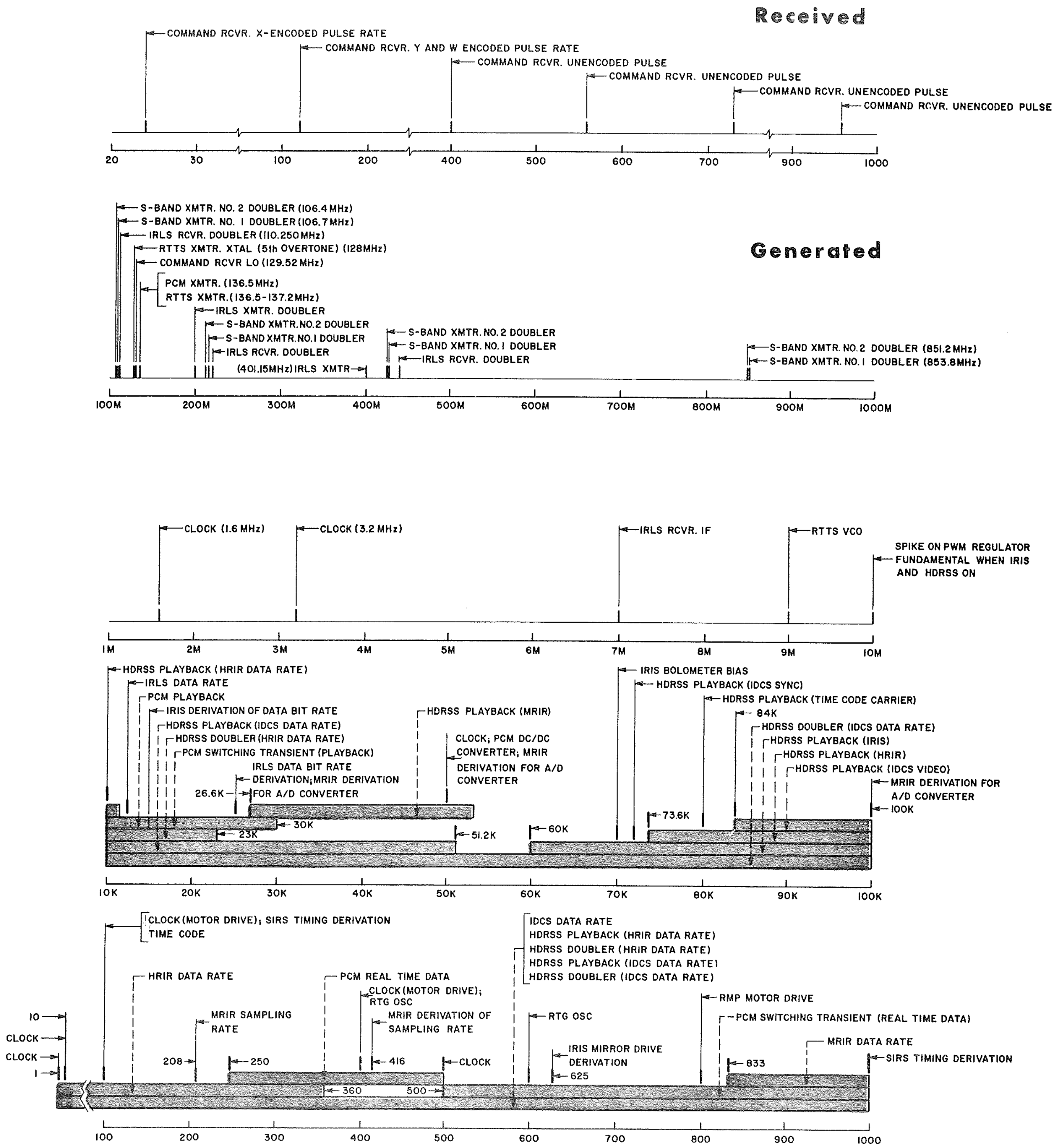
136.500	0.5W
1702.500	1.0W
1707.500	1.0W
2253.000	1.0W

Spurious Output Tests: For those experiments which generate internal frequencies or transmitter frequencies above 100 MHz in the normal mode of operation, tests shall be conducted to determine the power/frequency spectrum of such generators. These tests shall measure conducted RF energy at selected interface pins and radiated energy at specified points in the proximity of the experiment envelope. The energy measured at such points shall not exceed the limits specified by the following table:

Frequency Range	Attenuation (DB)
10 - 100 MHz	-40
.1 - 1 GHz	-60
1 - 10 GHz	-80
above 10 GHz	-140

Performance Degradation: Any degradation in Experiment performance while operating in the above RFI environment shall be reported to the Nimbus S/C Manager. The report must include the character and magnitude of the observed degradation and test environment. A decision will then be made upon what corrective action must be taken in either (or both) the experiment design or the spacecraft design.

NIMBUS B FREQUENCY SPECTRA



Frequency (Hz)

NIMBUS B FREQUENCY SPECTRA

RECEIVER FREQUENCIES (NON-RF)

S/S	FREQUENCY (10^{14} Hz)	TYPE OF RADIATION
IRIS	0.15-0.60(5-20μ)	INFRARED
SIRS	0.20-0.21(14.2-15μ)	INFRARED
MRIR	0.1-0.6; 0.75-15 (0.2-30μ)	ULTRAVIOLET, VISIBLE, IR
HRIR	0.71-0.88; 2.31-4.3 (0.7-4.2μ)	INFRARED
IDCS	4.0-6.25 (0.48-0.75μ)	VISIBLE
MUSE	10-300 (0.01-0.29μ)	ULTRAVIOLET

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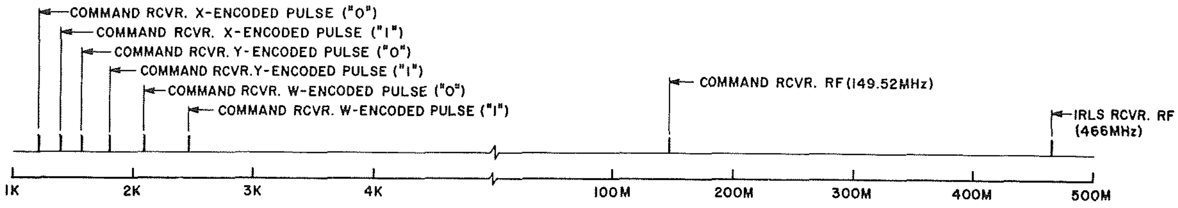

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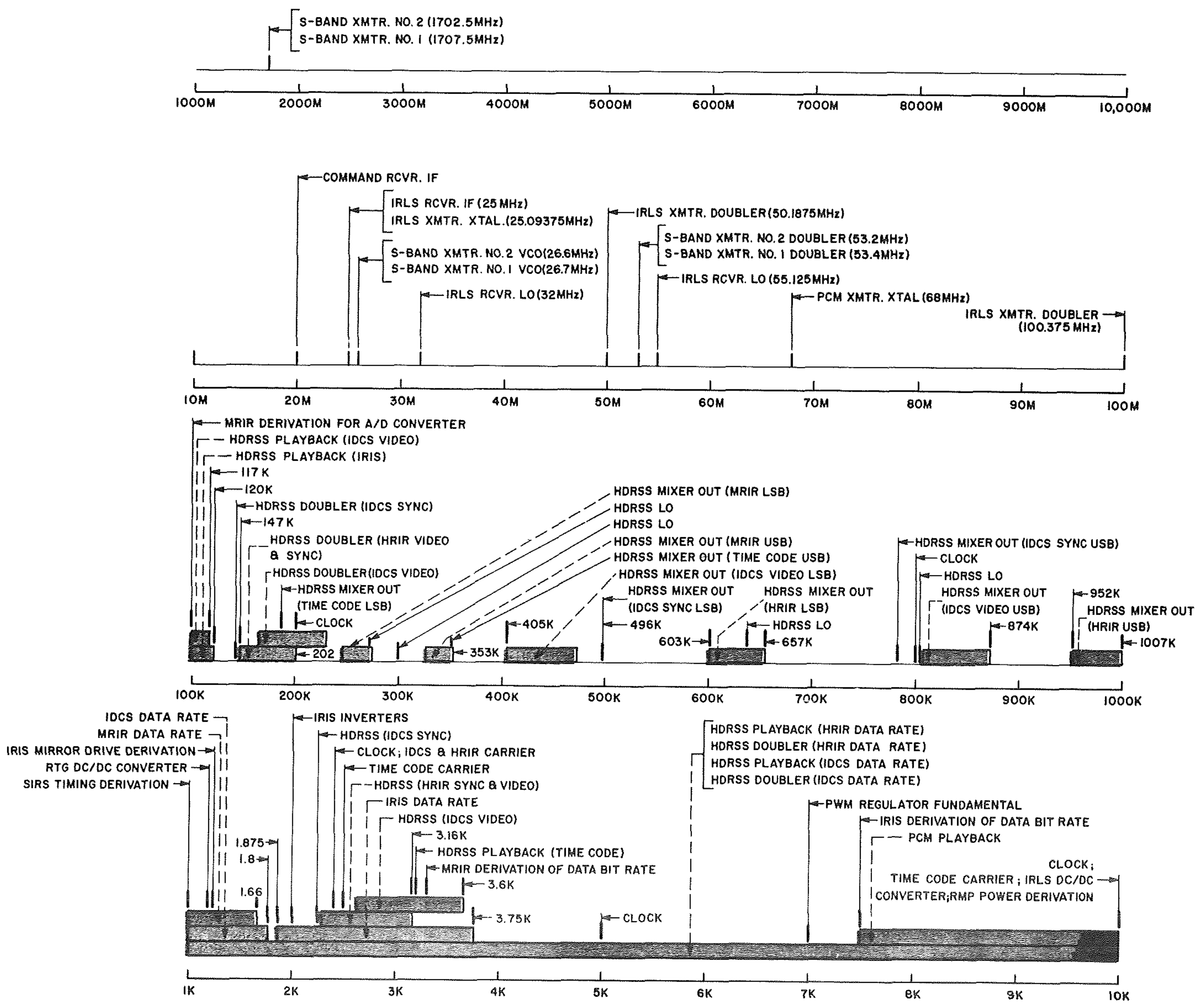


Frequency (Hz)

GROUNDING

Proper grounding effectively reduces the transmission by conduction of interfering energy (noise) from one system to another and within a system.

Grounding Philosophy

The grounding philosophy for NIMBUS E and F (see sketch below) is as follows:

SYSTEM UNIPOINT GROUND is established in close proximity to the power controller module and consists of a copper bar electrically and mechanically bonded to the spacecraft structure. Unipoint ground is the reference ground, and power ground from the power controller module is tied to this point through the power harness.

POWER GROUND is defined as a noise ground (choppers, solenoid switches, motors, relays, etc.) where the noise level is not detrimental to the operation of the subsystem. The power return should be twisted with the power lead.

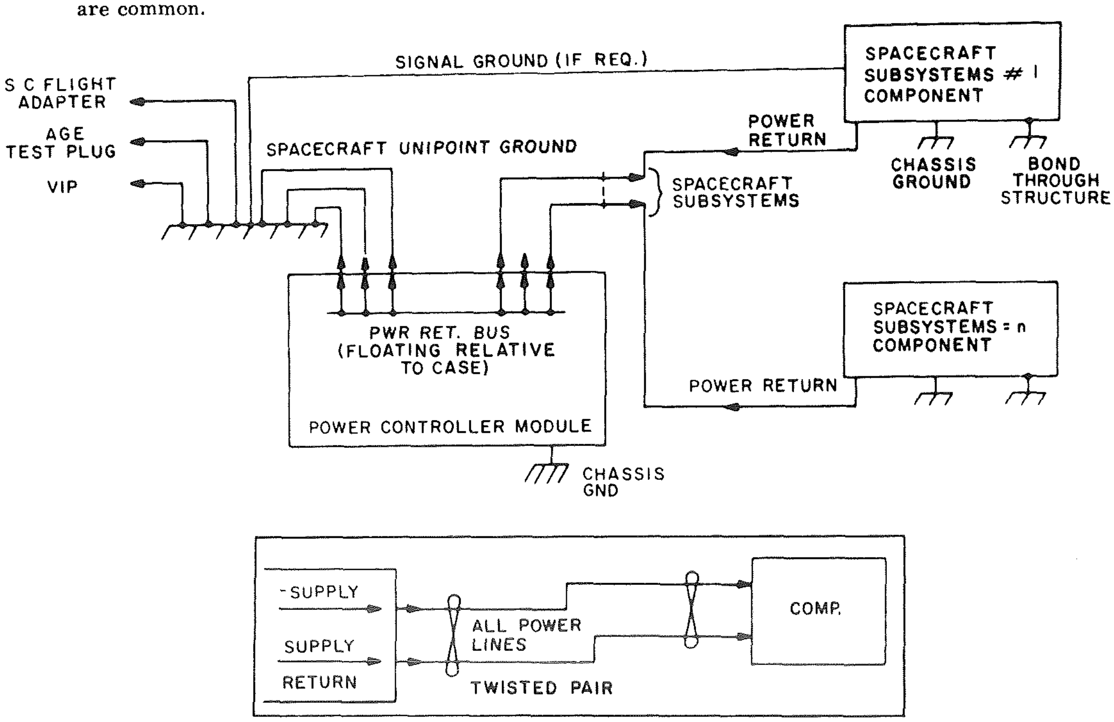
SIGNAL GROUND is used as a reference for low-level signals (video, error signals, sync signals, etc.). A low-level ground is a subsystem requirement because these circuits are susceptible to noise.

TELEMETRY GROUND is used as a reference for the low-level telemetry housekeeping and monitoring signals.

CHASSIS GROUND refers to the subsystem case or metallic structure and is accomplished by the following:

- BONDING of the chassis to the local structure, and
- A BACKUP ELECTRICAL PATH via wires attached to the chassis and the local structure which mate at a power connector pin for critical components.

AS A GENERAL RULE, the subsystem power, signal, and chassis grounds are isolated from each other. The subsystem grounds are wired separately to the unipoint ground at the spacecraft power supply. However, because of their electrical characteristics, certain subsystems (transmitters, receivers, etc.) use a unipoint ground within the system; namely, power, signal, and chassis grounds are common.



GROUND SYSTEM

Grounding Rules

- DC TO DC CONVERTERS must be used to isolate the signal ground from the power return.

  EACH CONNECTOR (except coax) must have two pins wired to signal ground and one to chassis ground.

  - THE SIGNAL GROUND PINS must be wired to the signal ground point in the experiment circuitry. Wiring to these pins will permit the spacecraft contractor to tie the experiment signal ground to a noise-free spacecraft ground if required.

  - THE CHASSIS GROUND PIN must be wired to the module chassis. The impedance from the connector pin to the chassis must be less than 5 milliohms. Shields may be tied to the chassis ground pin by the spacecraft contractor.

- POWER GROUND - At least two pins on the power connector must be wired to the spacecraft power return. If the current exceeds 3.0 amps, additional pins must be provided unless No. 16 connector pins and wires are used, in which case additional pins must be wired if the current exceeds 5.5 amps.

- SEPARATE POWER RETURNS - For telemetry monitors switched on and off, a separate power return must be provided and brought out to a connector pin on the power and VIP connectors.

- CLOCK - The isolation transformers used in the experiment for the clock interface are not to be grounded to either the power return or the chassis ground, but rather a separate connector pin is to be wired to each transformer for connecting to the low level signal ground at the spacecraft clock.

- COMMANDS - Command input leads (MA and MB lines) must NOT be grounded under any conditions.

- SHIELD GROUNDS - It is usually advisable to ground all shields internal to the module to the chassis and to bring the shields of wires interfacing at connectors out to connector pins (next to the signal lead connector pin). All internal component shield grounding must be brought to the attention of the NASA GSFC spacecraft manager and the spacecraft contractor so that the ground loops can be avoided when the spacecraft harness is connected.

SHIELDING

Proper shielding effectively reduces the transmission by radiation of interfering energy (noise) from one system to another and within a system.

Use of Shielding

- NOISY WIRING - Wire shielding will be used for all noisy wiring, clock signals, etc.) in the spacecraft harness in addition to shielding of bundles and physical separation from susceptible wiring (low-level signals, telemetry, data, etc.). Experimenters must therefore allocate a connector pin for the shield and determine if the shielding should be carried through the experiment package.

- SUSCEPTIBLE WIRING may also require shields in the spacecraft harness, depending upon the environment.

- ALL LOW-LEVEL SIGNAL LEADS (less than one volt peak amplitude) must be shielded and not brought out of the module.

Wire Shielding

- THE DEGREE OF ATTENUATION REQUIRED must be established. The type of shielding required can then be determined.

- THE SHIELD MAY CONSIST of braided wire or metal tubing.

  - BRAIDED WIRE SHIELDS must have at least four interwoven strands and provide coverage (shielding) of at least 92%.

  - MUST BE CONTINUOUS - The wire shield must be continuous along the wire path, inclusive of terminal blocks, junction boxes, and connectors.

    - NECESSARY HOLES in the shield must be kept small in area, the largest diameter being much smaller than the wavelength of the interfering frequency.

    - AT CONNECTORS - The wire shield must extend within the back of the connector. No more than one inch of unshielded wire shall protrude back of the connector pin or socket.

  - INSULATING SHEATH - Each wire shield must be covered with an insulating sheath to prevent electrical contact, except at the point of intentional bonding.

- USE OF TWISTED PAIRS AND COAXIAL CABLE also aids in reducing the transmission of noise by radiation by reducing the susceptibility of the cable to the noise.

  - TIGHTLY TWISTED - Twisted pairs (or triples) must be tightly twisted, with the distance along the wire required for a complete (360 degree) twist not to exceed five times the diameter of the wire plus insulation.

  - HIGH FREQUENCIES - If simple twisted pair is inadequate due to high frequency interference, the twisted pair must be shielded by an additional copper braid covering.

  - RF COAXIAL LINES - If a simple coaxial line proves inadequate, a multiple-shielded coaxial line must be used. In this event, the outer shield will not be used as a current-carrying conductor.

- SHIELDS MUST NOT BE USED as a return conductor (except for single shield coax), as current flowing on the surface of the shield can cause radiated RF energy.

- SPACECRAFT HARNESS SHIELDS - Shields carried through the spacecraft harness must be tied to a connector pin next to the pin for the conductor. Shield pins may be tied to the chassis ground pin by the spacecraft contractor.

Magnetic Shielding

For extreme magnetic shielding problems, multiple shielding application of nu-metal foils must be utilized.

- INNER SHIELD - Use a high permeability alloy such as conetic material for the inner shield.
- OUTER SHIELD - Use a high saturation alloy such as Netic material for the outer shield (to protect against external fields).
- INSULATING MEDIUM - Isolate the inner shield from the outer shield with an insulating medium.

Equipment Shielding

- ESTABLISH the degree of attenuation required. Weight and space considerations will then indicate the type of material to be used and the level of attenuation attainable.
- CLEAN AND MATCHED MATING SURFACES should be used.
- SHIELDING CONTINUITY must be maintained across mechanical discontinuities (pressure vent openings, access panels, etc.)
    - RF GASKETING should be used as a shield, if necessary, to improve the effectiveness of shielding between surfaces. Pressure applied to the gasket by mating surfaces shall be $\geq$ 20 psi. Pressure must not be greater than one-third the elastic limit of the gasket.
    - NECESSARY HOLES IN THE SHIELD must be kept small in area, the largest diameter being much smaller than the wave lengths of the interfering frequency.
    - LARGE HOLES, such as ventilating ducts, must be covered by a fine copper mesh. If this is not possible, RFI compensation should be accomplished by the installation of a wave guide attenuator over the holes.
- FOR THERMALLY ISOLATED EQUIPMENT OR STRUCTURAL ELEMENTS, the shielding path to structure shall be via a ground strap.
- USE OF AN ABSORPTION MATERIAL - Continual reflection within a shield may occur; thus a good shield may create an unwanted oscillating field. This can be corrected by the insertion of an absorption material.

Residual Magnetism

During the course of building, testing, and transporting a subsystem or experiment, it can be subjected to magnetic fields which could be damaging to the item. Experimenters must be aware of the magnetic environment and should know whether any such fields could cause degradation or damage to their units.

- VIBRATION EXCITER - One of the generators of magnetic fields is the vibration exciter.
    - TURN-ON AND TURN-OFF - The highest magnetic fields are generated during the initial turn-on and turn-off of the vibration system. It is strongly recommended that the shaker system be turned on (full operating current of the shaker field and degaussing system; ac power off) for at least 10 minutes prior to bringing the item near the shaker and that the item be removed from the shaker vicinity prior to shaker power shutdown.
    - PEAK FIELDS - Some of the older models can generate a field as great as 80 gauss during turn-on and turn-off and 30 gauss steady-state near the top of the exciter.
    - TYPICAL - Steady-state fields for shaker systems are typically 5 to 10 gauss.
- TRANSPORTATION - It has been reported by reputable authorities that exposures as high as 5 gauss have been recorded on packages during cross-country transportation.

FIELD EFFECT TRANSISTOR (FET) PRECAUTIONS

The input terminals of FETS and MOSFETS (field effect transistors and metal oxide semiconductor field effect transistors) must be brought out to connectors and protected by a shorting plug when not in use. The input terminals must be kept shorted when not in operation or undergoing test so that static charge cannot build up across the gate to body, punching through the dielectric.

Procedures

Procedures for installation and removal of the shorting plugs must be written and provided to the GSFC spacecraft manager and to the spacecraft contractor with the equipment. A typical procedure might consist of the following:

Do not remove shorting plugs except to run tests or attach to spacecraft wiring.

Use the following sequence for removal of shorting plugs.

Disconnect the shorting plug from connector pin __________ and install cable __________.

Disconnect the shorting plug from connector pin __________ (the other FET terminal) and install cable __________.

For reinstallation of shorting plugs, use the following sequence:

Disconnect cable __________ from connector pin __________ and install the shorting plug.

Disconnect cable __________ from connector pin __________ and install the shorting plug.

Additional Protection

The use of additional protective circuits must be investigated. If such circuits can be used they must be made a requirement, with documentation provided to the GSFC spacecraft manager. Such a circuit might consist of a Zener diode or a resistor to limit the voltage or bleed off the charge, respectively, across the input terminals.

Section VI

TEST PROGRAM

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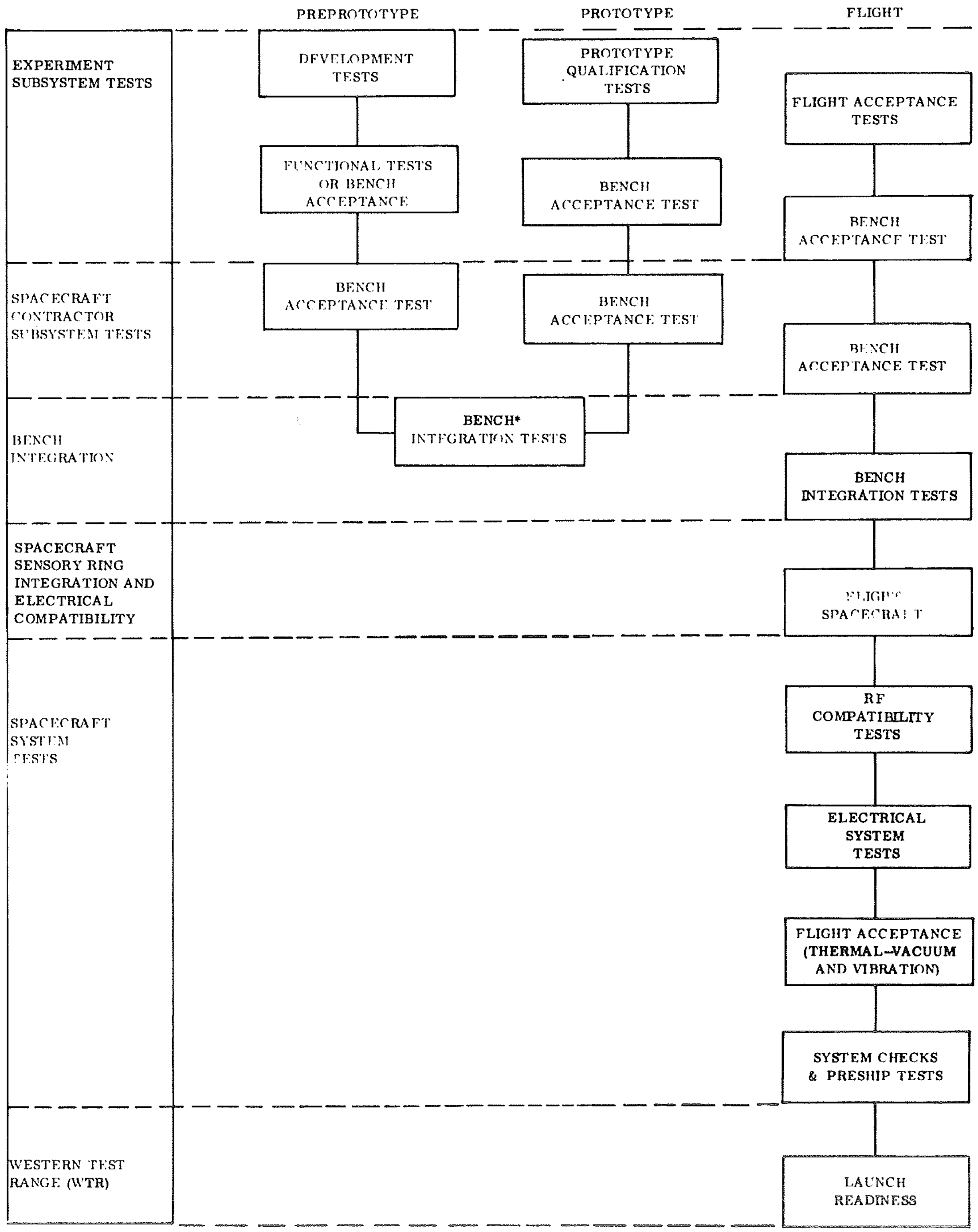
TEST FLOW DIAGRAM

The diagram below shows the flow of hardware from subsystem level testing through integration and spacecraft system testing to prelaunch testing at the Western Test Range.



*ENGINEERING MODEL OR PREPROTOTYPE HARDWARE INSTALLED ON THE BIT BOARD IS REPLACED BY PROTOTYPE HARDWARE AS IT BECOMES AVAILABLE

EXPERIMENT TESTING

Definition of Models

- ENGINEERING (PREPROTOTYPE) MODEL HARDWARE

  Preprototype model hardware must have the same mechanical and electrical interface and dimensions as the flight hardware, but they need not use preconditioned components and must not contain potting required for structural resistance to vibration. The engineering model must be capable of operating in a thermal-vacuum chamber from $10^{o}C$ to $40^{o}C$ for a gross checkout of chamber facilities involving experiment thermal-vacuum calibration targets, chamber interface wiring, spacecraft thermal-vacuum test procedures, etc.

- PROTOTYPE MODEL HARDWARE

  Prototype model hardware must be identical to flight hardware design and must have been successfully tested to all applicable specification requirements. In addition, prototype models must be qualified to the prototype environmental levels as delineated in the applicable GSFC specification (see Test Environments Section).

- FLIGHT MODEL HARDWARE

  Flight model hardware must be identical in all respects to the prototype model which was successfully qualification tested. In addition, flight hardware must be acceptance tested to the flight environmental levels as delineated in the applicable GSFC specification. These levels are somewhat less severe than prototype levels.

Subsystem (Experiment) Level Tests

The experimenter is responsible for performing all subsystem environmental tests and electrical tests to applicable contract specifications. All records of testing and calibrations must accompany the delivery of equipment to the spacecraft contractor facility. In addition, the hardware developed for electrical checkout (bench checkout units) must accompany delivery of models to the spacecraft contractor facility, as it must be used during bench acceptance and bench integration tests.

Summary of Subsystem Testing

The following is a summary of tests to be conducted by experimenters prior to shipment of hardware to the spacecraft contractor. Experiment contracts and specifications will define specific requirements, however, the following is generally applicable.

- ENGINEERING (PREPROTOTYPE) MODELS - Development functional tests; limited environmental tests; bench acceptance test if bench checkout equipment is available at the time.

- PROTOTYPE MODELS - Qualification tests to applicable specifications; bench acceptance test prior to shipment.

- FLIGHT MODELS - Flight acceptance tests to applicable specifications; bench acceptance test prior to shipment.

Bench Acceptance Tests

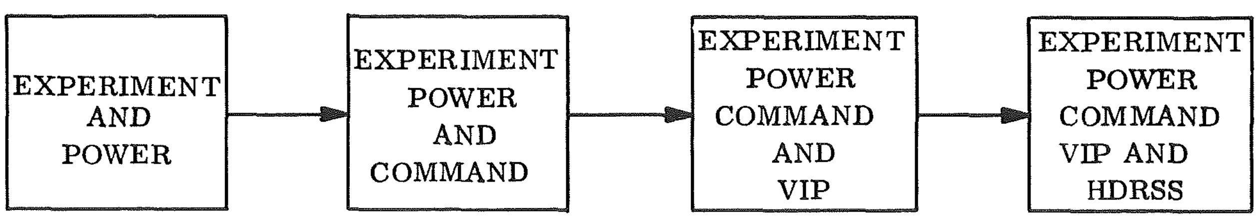
Bench Acceptance tests are the final tests performed by the experimenter prior to hardware delivery to the spacecraft contractor and the first tests performed on the experiment by the spacecraft contractor. The purpose of these tests is to demonstrate experiment performance prior to shipment and to verify performance after delivery to the spacecraft contractor facility. Bench acceptance testing is intended to allow maximum accessibility to the experiment for checking its operability and compatibility with simulated spacecraft interfaces. Therefore, it is in the experimenter's best interests to develop complete, accurate bench acceptance test procedures and Bench Checkout Units (BCU).

- BENCH ACCEPTANCE TEST PROCEDURES must be reviewed and approved by both the experimenter and the GSFC spacecraft manager to ensure that both the experiment design and its compatibility to the spacecraft interface are verified. Spacecraft contractor review of Bench Acceptance Test Procedures is also required.

  BENCH CHECKOUT UNITS must include both hardware to verify the experiment design and simulated spacecraft subsystem interfaces for commands, telemetry, power, data handling, and any other electrical interfaces which are applicable. BCUs must also include BCU targets and an appropriate interface for the experiment test points. BCU targets are also used as part of the spacecraft check-of-calibration adapter--used in the calibration test at ambient--and must therefore be compatible with that adapter.

Bench Integration Tests

The spacecraft contractor performs a Bench Integration Test Program to integrate each experiment with all other subsystems with which it interfaces. The purpose of this program is to reveal early in the Program any basic incompatibilities in the operation between subsystems, performance anomalies, or errors in hardware design or fabrication prior to installation in a spacecraft. Where possible, subsystems are operated with their ground stations to further assure system compatibility. Bench integration test requirements and procedures are developed from design documentation, performance requirements, test reports, and operating and handling instructions provided by experimenters. During a typical bench integration test (see diagram below) an experiment is integrated with the power, command, HDRSS and VIP subsystems. When all experiments are integrated, a series of "system" level tests is performed to evaluate experiment compatibility with other subsystems, to checkout system test procedures, and to confirm ground station hardware and software performance.



TYPICAL BENCH INTEGRATION TEST FLOW

SPACECRAFT TESTING

After Bench Acceptance and Bench Integration testing (see previous section, "Experiment Testing), the spacecraft contractor initiates the integration of experiments and spacecraft subsystems into the sensory ring. The following summarizes the testing performed from sensory ring integration through prelaunch tests of the flight spacecraft at the Western Test Range.

- SENSORY RING INTEGRATION - Experiment and subsystem hardware is installed in the spacecraft sensory ring and checked for compatibility with the command, clock, VIP, HDRSS, and power interfaces. Various power transient and noise measurements are taken. Mechanical checks performed include a mechanical fit check and installation and checkout of spacecraft harnessing.

- ELECTRICAL COMPATIBILITY - This test series is a completion of the sensory ring integration test using the assembled spacecraft. Its duration is the equivalent of approximately four orbits. Compatibility in terms of conducted electrical energy is determined by turning a subsystem or experiment on, operating it, and turning it off, while monitoring the effects at various points in the system. Compatibility with the ground station and the AGE is also verified.

- RF COMPATIBILITY - This test series is performed in the spacecraft contractor anechoic facility to detect interferences due to radiated electrical energy. The effects of transmitters upon subsystems, experiments, and particularly upon receivers are determined by the running of three test orbits.

- ELECTRICAL SYSTEMS - The electrical systems tests are the final complete electrical checkout of the spacecraft. Calibrations and bench mark telemetry levels become the baselines used in thermal-vacuum testing and checkouts at the launch site.

- THERMAL-VACUUM - The thermal-vacuum test on the flight spacecraft includes electrical operation verified at each temperature plateau. Calibration and telemetry levels are recorded on the flight spacecraft for verification of in-orbit performance.

- VIBRATION

  - PROTOTYPE LEVEL TESTS, if conducted, are run on an optional spacecraft model (mechanical test model), which will be used specifically for qualification of the deployment mechanism.

  - FLIGHT ACCEPTANCE LEVEL TESTS are performed on the Flight Spacecraft.

- FINAL SYSTEM CHECKS AND PRESHIP TESTS - Included are such tests as alignment, spring determination, solar paddle check, antenna tuning, interface checks, pneumatic leak tests and workmanship vibration.

Prelaunch Tests

- THE FLIGHT SPACECRAFT undergoes prelaunch tests to verify operability and overall system launch readiness.

- PRELAUNCH TESTING is conducted in two areas, the Satellite Assembly Building (SAB) upon arriving at the WTR and at the launch pad. The diagram below outlines the spacecraft testing performed at WTR at these two locations.

SAB TESTS

FLIGHT SPACECRAFT

- Post-Ship Inspection
- Battery Charge/Capacity Check
- Confidence Test
- Spacecraft Alignment
- Solar Paddle Continuity
- Leak Test
- Battery Conditioning
- Confidence Test (Repeat)
- Pneumatic Charge

PAD TESTS

- Mate with AGENA Shroud Ring
- Dry Air Purge
- Adapter I/F connections
- Visual Inspection
- S/C Functional Check
- Shroud Installation
- Clearance Tests
- RF Link Test
- S/C Confidence Test
- RF Link Check
- S/C Go/No-Go Check
- Mock Countdown
- Arm Pyrotechnics
- S/C Confidence Test
- Countdown

TEST ENVIRONMENTS

Test environments for NIMBUS E and F subsystems and experiments will be delineated in the applicable GSFC specification. Levels to be tested may vary in some cases with location of the equipment in the spacecraft (i.e., bay mounted, center section mounted, below the sensory ring, etc.). The values summarized below are offered only as design guidelines.

Prototype Level (Qualification) Test Environments

The values given are worst case and are only to be used as design guidelines.

- VIBRATION

  Sinusoidal: 5 to 2000 cps, 10 G (0 to peak) all axes, sweep rate 1 octave/minute
  Random: 20 to 2000 cps, 20 G-RMS all axes, 4 minutes/axis

- ACCELERATION: 30 G for 5 minutes

- THERMAL-VACUUM: Temperature cycling between -5 and 45°C at a pressure of less than 10^{-5} mm Hg

- HUMIDITY: 95% relative humidity at 30°C for 24 hours

Flight Level (Acceptance) Test Environments

These environments are typically somewhat less than prototype levels, and are intended to simulate the actual operating environment. The values given are worst case and are only to be used as design guidelines.

- VIBRATION:

  Sinusoidal: 5 to 2000 cps, 5 G (0 to peak) all axes, sweep rate 2 octaves/minute
  Random: 20 to 2000 cps, 11.7 G-RMS all axes, 2 minutes/axis

- THERMAL-VACUUM: Temperature cycling between 0 and 40°C at a pressure of less than 10^{-5} mm Hg.

Component Operation During Testing

Components are operated during tests in the manner in which they would be operating when encountering the applicable environment. For example:

- VIBRATION AND ACCELERATION

  Components are operated in the same manner as during launch lift-off and ascent for vibration and acceleration testing.

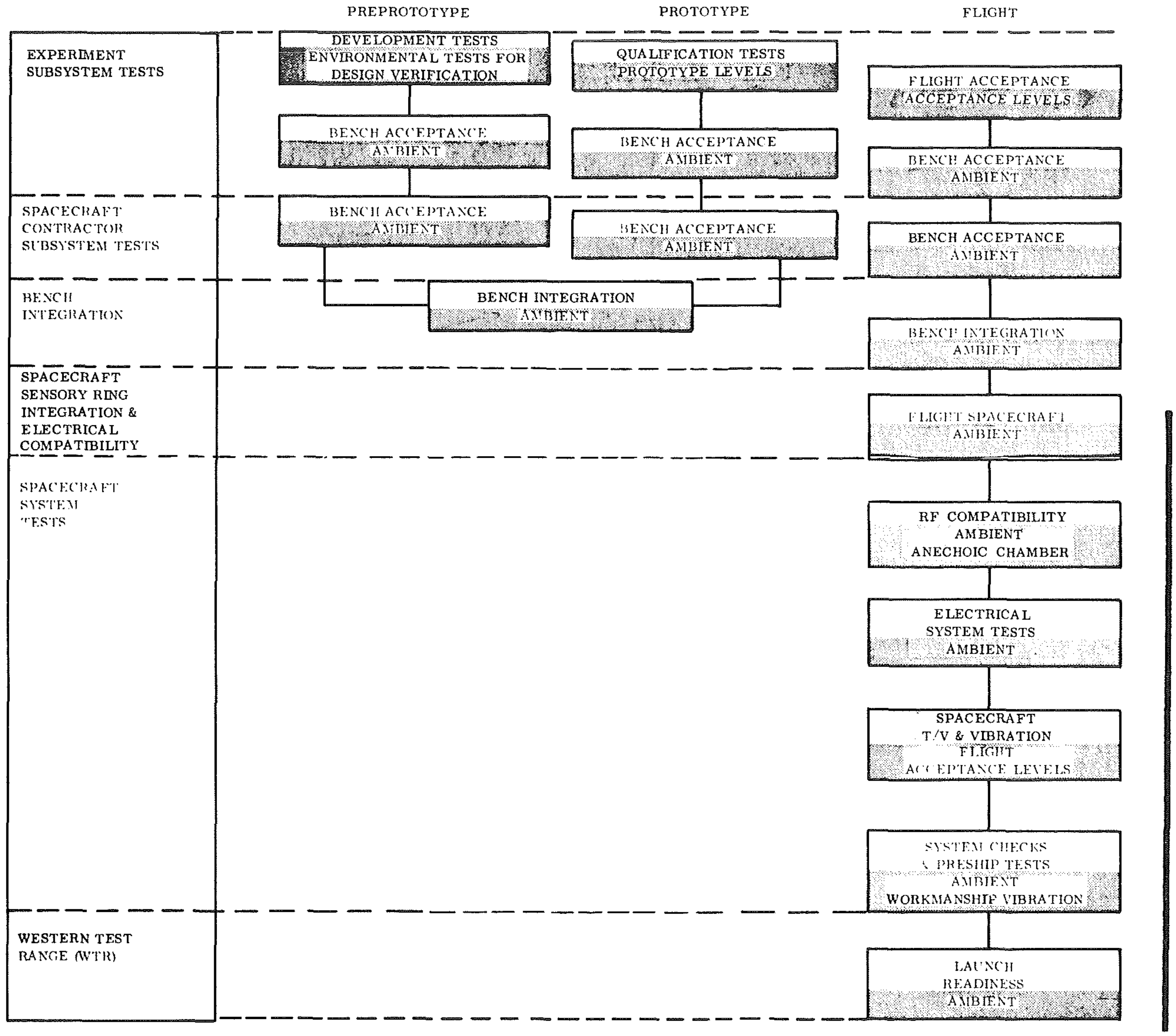
- THERMAL-VACUUM

  Components are operated in the normal in-orbit mode during thermal-vacuum testing.

- HUMIDITY

  Components are subjected to high relative humidities only during storage and check-out prior to launch, so components must be capable only of surviving (nonoperating) the humidity testing and satisfactory operation thereafter.

TEST PROGRAM ENVIRONMENTS



TEST DATA

The handling and availability of test data are limited by the test data link and equipment. The summary below and the table on the opposite page outline the major tests and describe their data links and primary sources of test data.

Bench Acceptance

The experimenter provides the harnessing, test equipment, and stimuli for bench testing his experiment to assure satisfactory performance. The test equipment (Bench Checkout Unit) is the primary source of test data.

Bench Integration

The spacecraft contractor provides inter-subsystem harnessing and, in conjunction with bench acceptance harnessing and equipment provided by experimenters, performs experiment/subsystem compatibility tests. BCU, telemetry and experiment data are provided.

Sensory Ring Integration and Electrical Compatibility

In this test series, the various subsystems are tested individually and are then progressively tied together in a spacecraft sensory ring to check overall systems performance. As the subsystems are tied together, telemetry and experiment data become the primary method available to determine subsystem performance.

Electrical Systems and RF Compatibility

During the complete electrical checkout of the spacecraft and operating tests for RF compatibility, telemetry and experiment data make up the primary test data.

Thermal-Vacuum

When the spacecraft is placed in the thermal-vacuum chamber, testing becomes more restrictive because there is no physical access to the spacecraft. Spacecraft telemetry and experiment data are hard-wired out of the chamber and processed by the appropriate ground station computer. The ground station computer can be programmed to formats to suit the experiment requirements. The experimenter provides calibration targets which are mounted in the thermal vacuum check-of-calibration adapter and used to stimulate sensors to provide quantitative, though limited, data.

Launch Site Testing

At the launch site, testing is restrictive. There is no physical access to the spacecraft except at the Satellite Assembly Building. Once on the Pad, no further experiment checkout is performed. Only experiment and telemetry data are available.

Major Test Factors

The length of the test program and various restrictions outlined above point up the importance of many factors, the most significant of which are:

- EXPERIMENT HARDWARE RELIABILITY - Failure of any component in spacecraft environmental test requires retest, which means all equipment may be subjected to testing many times.
- BENCH CHECKOUT EQUIPMENT must be reliable and adequate for experiment checkout.
- EXPERIMENT TARGETS must be capable of performance and control in ranges and tolerances consistent with experiment ranges and tolerances.
- TELEMETRY MONITORS must be carefully selected consistent with the limitations of access to test data defined above.

TEST DATA LINKS and OUTPUTS

Test	Test Objectives	Data Link	Primary Data Source(s)
Bench Acceptance	Verify experiment operability	Hardwire	BCU data
Bench Integration	1. Check subsystem electrical interface compatibility 2. Bench mark telemetry levels	Hardwire, VIP and HDRSS	BCU, telemetry and experiment data
Sensory Ring Integration and Electrical Compatibility	1. Check mechanical and electrical compatibility of the subsystem in the Sensory Ring: a. Mechanical Fit Check b. Mate-to-Harnessing c. How subsystem works in conjunction with other subsystems and the spacecraft. 2. Check spacecraft and ground station compatibility 3. Check spacecraft and AGE compatibility.	VIP, HDRSS, (hardwire)	Telemetry, experiment and some BCU data
RF Compatibility	1. Check how transmitters affect operations and data. 2. Perform air link sensitivity tests. 3. Perform RF measurements.	VIP and HDRSS (open air)	Telemetry and experiment data, and some additional RF measurements
Electrical Systems	1. Final complete electrical checkout of the spacecraft 2. Determine calibration and bench mark telemetry levels to be used as baselines for thermal-vacuum testing and tests at WTR	VIP and HDRSS (hardwire)	Telemetry and experiment data
Thermal-Vacuum	1. Calibrate subsystems to calibration targets. 2. Determine flight performance baselines (note signatures present in T/V environment) 3. Check electromagnetic compatibility (EMC) 4. Perform Leak Tests	VIP, HDRSS and hardwire	Telemetry and experiment data, and some additional spacecraft and calibration target temperatures
Vibration	1. Determine spacecraft performance during and after vibration environment	VIP and HDRSS (open air)	Telemetry and experiment data
Prelaunch/Launch	Verify Spacecraft Operability	VIP and HDRSS (open air and hardwire)	Telemetry and experiment data

Section VII

MANAGEMENT

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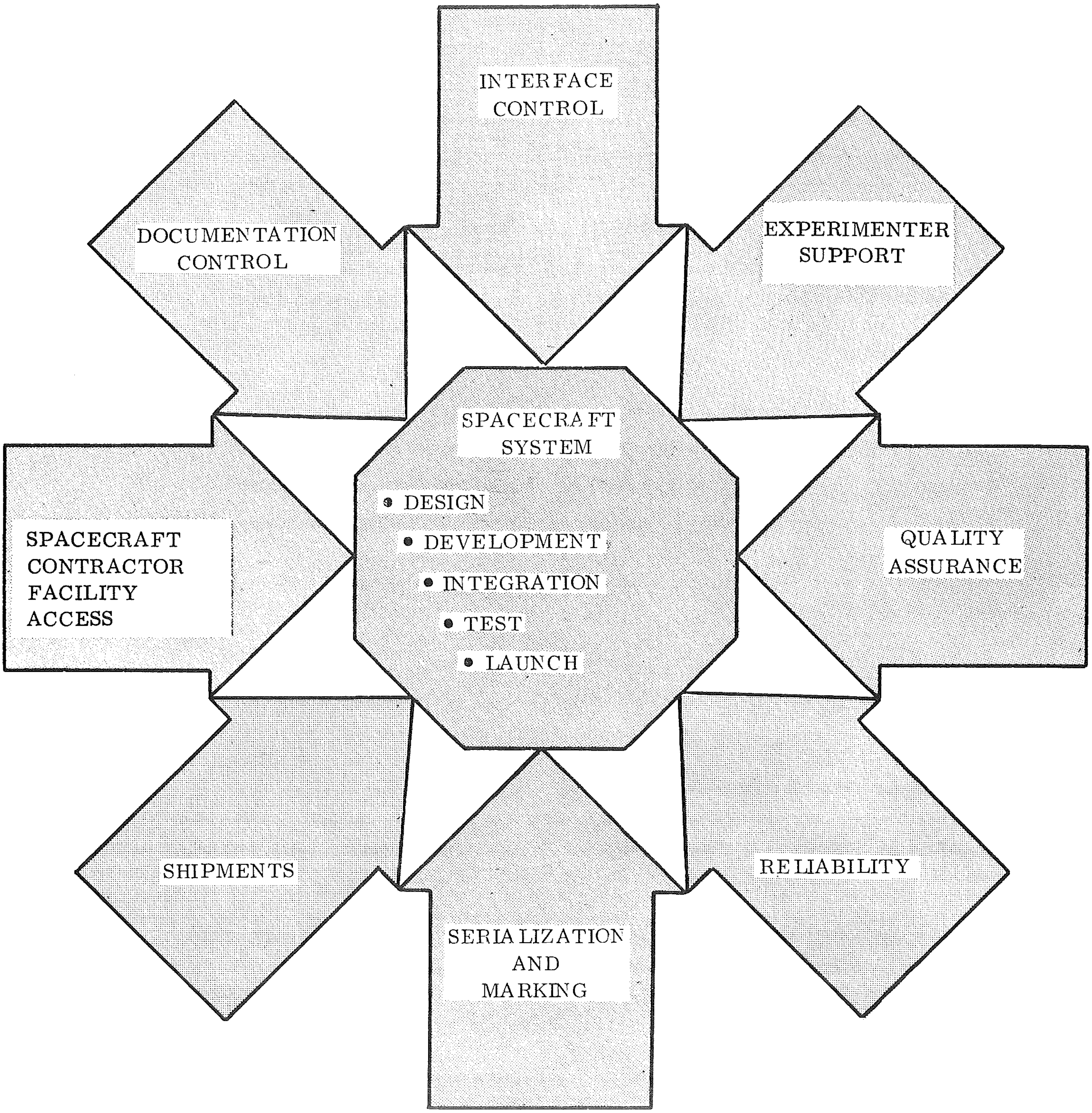
MANAGEMENT INTERFACE



INTERFACE CONTROL

INTERFACE CONTROL SHALL BE ATTAINED THROUGH:

- Establishment of clear lines of communications between NASA, experimenters and participating contractors
- Scheduled interface meetings
- Documentation of requirements and interface design
- Change control of interface documentation

Cognizant NASA and contractor representatives will be identified for each experiment subsystem, the spacecraft subsystems and system to assure the proper distribution of information and establishment of personal contacts for technical interchange.

Interface Meetings

Interface meetings will be scheduled on a monthly basis (with special meetings more frequently as determined necessary) until that time when interface design is firm. Interface meetings will be attended by experimenters and cognizant NASA and contractor representatives. The purpose of interface meetings will be to establish compatibility between the experiments and the spacecraft as early as possible through:

- DEFINITION AND RESOLUTION of actual and potential design incompatibilities.
- DOCUMENTATION of detail interface requirements and design as they are developed.
- FAMILIARIZATION OF EXPERIMENTERS with spacecraft design and performance characteristics.
- FAMILIARIZATION OF SPACECRAFT CONTRACTOR PERSONNEL with experiment design and performance characteristics.

A uniform agenda will be used for all interface meetings, and the documentation of minutes and action items will be required. The basic agenda will be as follows:

- Action Item Review
- Experiment Design Status
- Problems
- Interface Agreement Status
- Action Items

Interface Control Documentation

The spacecraft contractor will prepare and publish interface control documentation utilizing NASA and experimenter inputs, experiment drawings, results of interface meetings and spacecraft system design information. Interface control documentation will include:

- Interface agreements (the controlling document)
- Mechanical interface drawings
- Electrical interconnect tables
- Thermal drawings.

These documents are the mechanism for early definition of all interfaces and for protecting interface design from unilateral change.

Change Control

Timely interchange of interface information during the early stages of the program is mandatory and will be facilitated by informal control of preliminary issues of interface documents. Formally released interface agreements and subsequent revisions shall require approval by NASA, experimenters and cognizant contractors by document sign-off. Changes required during the program development should be requested through the cognizant NASA technical officer, who in turn will request the NASA spacecraft manager to call a special interface meeting or include the change on the agenda of the next scheduled meeting for assessment and negotiation. Thus, system considerations and trade-offs, as well as experiment considerations, can be applied to reach the decision which best satisfies overall program requirements.

INTERFACE AGREEMENTS

EACH EXPERIMENTER WILL BE REQUIRED TO SUPPORT WITH TECHNICAL INPUTS THE PREPARATION OF AN INTERFACE AGREEMENT. THE BASIC FORMAT FOR ALL INTERFACE AGREEMENTS WILL BE AS FOLLOWS:

Scope

Applicable Documents

Interfaces

ELECTRICAL INTERFACE

- System Schematic
  - Signal Characteristics Table
  - Signal Flow Information
- Power Requirements
  - Power Allotment
  - Thermal Dissipation
  - Power Profile
  - Spacecraft Power Regulation
  - Regulated Bus Transients
  - Power Supply Input Circuits
  - Power Status Reporting
- Command and Clock Requirements
  - Command Signal Interface Definition
  - Standard Frequency Signal Interface Definition
  - High Level Motor Drive Signals
  - Modulated Time Code Signals
- Telemetry Requirements
  - Telemetry List
- VIP Interface Information
  - Analog Input Interfaces
  - Digital "A" Input Interfaces
  - Digital "B" Input Interfaces
  - Major Frame Pulse Interfaces
- Interfaces with Other Subsystems (or Experiments)

MECHANICAL INTERFACE

- Configuration - Mechanical Interface Drawing
- Mass Properties
  - Weights (Estimated, Calculated, or Actual)
  - Center of Gravity
  - Moments of Inertia
- Mounting Definition
  - Alignment
  - Thermal Surfaces
  - Torquing Instructions
- Placement Restraints

ENVIRONMENTAL INTERFACE

- Temperature
- Deviations from Specification
- Humidity Control
- Contamination
  - Outgassing
  - Adherence
  - Internal Cleanliness

Operational Restraints

- Safety Precautions
- Access
- Separation
- Test
  - Subsystem Test
  - System Test
- Handling
- Red Line Restraints (Allowable Operating Times)

Documentation of Hardware

- Hardware
  - Experiment Models
  - Bench Checkout Equipment
  - Targets
  - Other Hardware
- Documentation
  - Schematics
  - Signal Flow Diagrams
  - Bench Check Procedures
  - Log Books
  - Connector Mate/Demate History
  - Instruction Manuals with Telemetry Calibration

Interface Change Control Procedure

EXPERIMENTER SUPPORT

EXPERIMENT PERFORMANCE IS THE RESPONSIBILITY OF THE EXPERIMENTER. It is therefore his responsibility to provide the necessary support in equipment and personnel during the spacecraft design, integration, test and prelaunch phases.

Suitable Equipment

Each experimenter must provide suitable equipment to permit:

- Incoming test and inspection of experiment hardware at the spacecraft contractor facility.
- Suitable tests during bench integration with the spacecraft subsystems and other experiments.
- Checkout of the experiment during spacecraft integration, test, and qualification.
- Effective system confidence tests and prelaunch evaluation.

The equipment must include the following:

- BENCH CHECKOUT UNITS *Test support equipment which simulates spacecraft interfaces and* enables the complete electrical checkout of the experiment.
- TARGETS - Suitable targets are required during experiment evaluation and prelaunch confidence tests (Must be controllable to tolerances consistent with the function of the test and the equipment being tested.) They must include at least the following:
  - BCU TARGETS are used for bench check tests at ambient, and as part of the check of calibration at ambient. BCU targets must therefore be compatible with the spacecraft check-of-calibration adapter.
  - THERMAL-VACUUM TARGETS are used in functional checkouts during thermal-vacuum testing.
  - GO-NO GO TARGETS used on previous NIMBUS spacecraft in prelaunch confidence tests are not planned for use on NIMBUS E or F.

Personnel

- RESIDENT

  A full time resident at the spacecraft contractor facility may be required for any experiment. Typical resident responsibilities would include supporting all calibration testing, maintenance of experimenter - provided calibration targets, support for special checkout equipment, and general engineering support.

- OTHER PERSONNEL

  Upon the determination of need by the NASA spacecraft manager, the experimenter may be required to provide the following support personnel:

  - TECHNICAL SUPPORT at the spacecraft contractor facility, the launch site, and possibly special support when test problems occur or critical tests are being performed.
  - FACTORY SUPPORT for repairs or modifications of equipment.

Summary of Experimenter Responsibilities

The experimenter or his designated representative must:

- CERTIFY EXPERIMENT PERFORMANCE to specifications upon delivery to the spacecraft contractor facility.
- PARTICIPATE IN ESTABLISHING AND CONTROLLING THE INTERFACE between his experiment and the spacecraft through participation in the development and control of interface design, interface agreements, and mechanical and electrical interface drawings.
- ESTABLISH AND PROVIDE detailed bench acceptance test procedures which define the step-by-step procedure for performance of the test, special precautions, and test outputs or success criteria.
- PROVIDE 24 HOUR SUPPORT OF TESTING, as required, during the bench acceptance test, bench integration test, subsystem assembly, integration, spacecraft system test and prelaunch phases.
- ARRANGE FOR THE MAINTENANCE AND REPAIR of his experiment and for general factory support of the integration effort (on a crash basis if necessary).
- REVIEW SECTIONS OF DETAILED SYSTEM/SUBSYSTEM TEST PLANS that affect his experiment.
- EVALUATE DATA relative to his experiment in support of the spacecraft contractor during development and key system and subsystem tests.
- HELP IN TROUBLESHOOTING PROBLEMS relating to his experiment.
- PROVIDE ANY SPECIAL PURPOSE EQUIPMENT necessary to evaluate the experiment.
- ARRANGE FOR TRAINING of resident field service personnel who will operate support equipment, and assist in maintaining this equipment (ground station and bench checkout equipment)
- PROVIDE PRELAUNCH EVALUATION of his experiment at WTR, and inform the NASA spacecraft manager when his experiment is ready for launch.

Experiment Facilities

The experimenter must ensure that facilities and materials are available if rapid repair, updating, and checkout of the equipment become necessary during the program. A rapid turn-around of equipment repair and updating is needed to maintain program schedule requirements.

DOCUMENTATION CONTROL and AVAILABILITY

Library

A program library will be established at the spacecraft contractor's integrated test facility. The library will have in storage all spacecraft documentation, interface documentation, and experiment subsystem documentation. These documents will be available to NASA, experimenter and contractor personnel and will be located in the test facility to make them available during integration and system testing. To ensure that this library has complete documentation, experimenters shall provide to the spacecraft contractor complete sets of the following documents (as they become available):

- Drawings
- Specifications
- Test procedures
- Test reports
- Operation and maintenance manuals

Progress Reports

Three complete copies of all experimenter progress reports must be sent to the NASA GSFC spacecraft manager for use in program evaluation, planning and schedule integration. Information of importance to NASA and the spacecraft contractor which should be included in monthly reports:

- Program status against milestones
- Projected schedule and changes, particularly acceptance test and delivery schedules.
- Additions/changes to the List of Materials
- Design or performance changes
- General test information (successes and failures)

Failure Reporting

Failure reporting at the spacecraft contractor's facility with respect to experiment subsystem hardware is accomplished as follows:

- INSPECTION MALFUNCTION REPORT

  Upon incoming inspection of experiment hardware, a GSFC malfunction report will be completed defining all deviations from the interface agreement and drawing and omissions of required documentation. The reports will be forwarded to the NASA-GSFC quality assurance representative, who will in turn require disposition by the experimenter through the cognizant NASA technical officer. It is important that timely disposition be made of all deviations, especially in the case of hardware deviations from drawings.

- INTEGRATION & TEST MALFUNCTION REPORT

  During integration and test, a GSFC malfunction report will be completed and handled as stated above. However, during this phase of the program, rapid disposition may be necessitated to avert undue delays. A special procedure will be implemented by NASA to handle such cases.

QUALITY ASSURANCE and RELIABILITY

The following documents define the quality assurance and reliability requirements implemented on the NIMBUS program. The applicability of these documents and their requirements to experiment subsystems is as specified in each experiment subsystem contract. It is required that all experimenters effect programs which implement applicable requirements to assure the high, uniform quality and reliability required for successful NIMBUS E and F system performance.

Quality and Reliability Assurance Specifications

1. GSFC Specification, "Quality and Reliability Provisions for NIMBUS E and F Experiments", NASA Publication S-450-P-9 dated 1 Oct. 1968.
2. "Preferred Parts List", GSFC Document PPL 10, dated 1 July 1968.

Additional QA&R Documents

- QUALITY ASSURANCE AND RELIABILITY PROGRAM PLANS must be prepared, submitted for NASA approval, and carefully carried out.
- QUALITY LOG BOOKS, INSPECTION AND TEST RECORDS, AND EQUIPMENT OPERATING TIME RECORDS MUST ACCOMPANY THE HARDWARE to the spacecraft contractor's facility. These data are necessary for maintaining overall spacecraft quality control and for reference during subsequent testing.

SERIALIZATION and MARKING

Nameplate Designation

All experiment hardware must be serialized by placing the appropriate designation on the nameplate. Nameplates are required for all preprototype (engineering models), prototype, and flight hardware, which are to be designated as follows:

PP = Preprototype - Electrically and mechanically like flight hardware, but required only to operate in an ambient environment.

PR = Prototype - Identical to flight hardware, but used for qualification test and spacecraft integration testing.

FT = Flight - Intended for flight use only.

FS = Flight Spare

<u>FORMAT OF SERIALIZATION</u>

```
┌Program - Nimbus (N)
│      ┌Spacecraft (E or F)
│      │     ┌Type of Unit (PP, PR, FT, FS)
│      │     │        ┌Unit Number (1, 2, 3 etc.)
│      │     │        │
X      X    XX        X
```

<u>Examples</u>

a.	The second of two Nimbus E Preprototypes	NE-PP2
b.	Nimbus F Flight Unit	NF-FT1
c.	Nimbus E Spare	NE-FS1

Support Hardware

<u>ALL SUPPORT HARDWARE MUST BE MARKED OR TAGGED</u> to assure against loss and to facilitate its usage. Test cables should be tagged, numbered and identified with the experiment hardware. The same applies to bench test equipment and miscellaneous test and support equipment

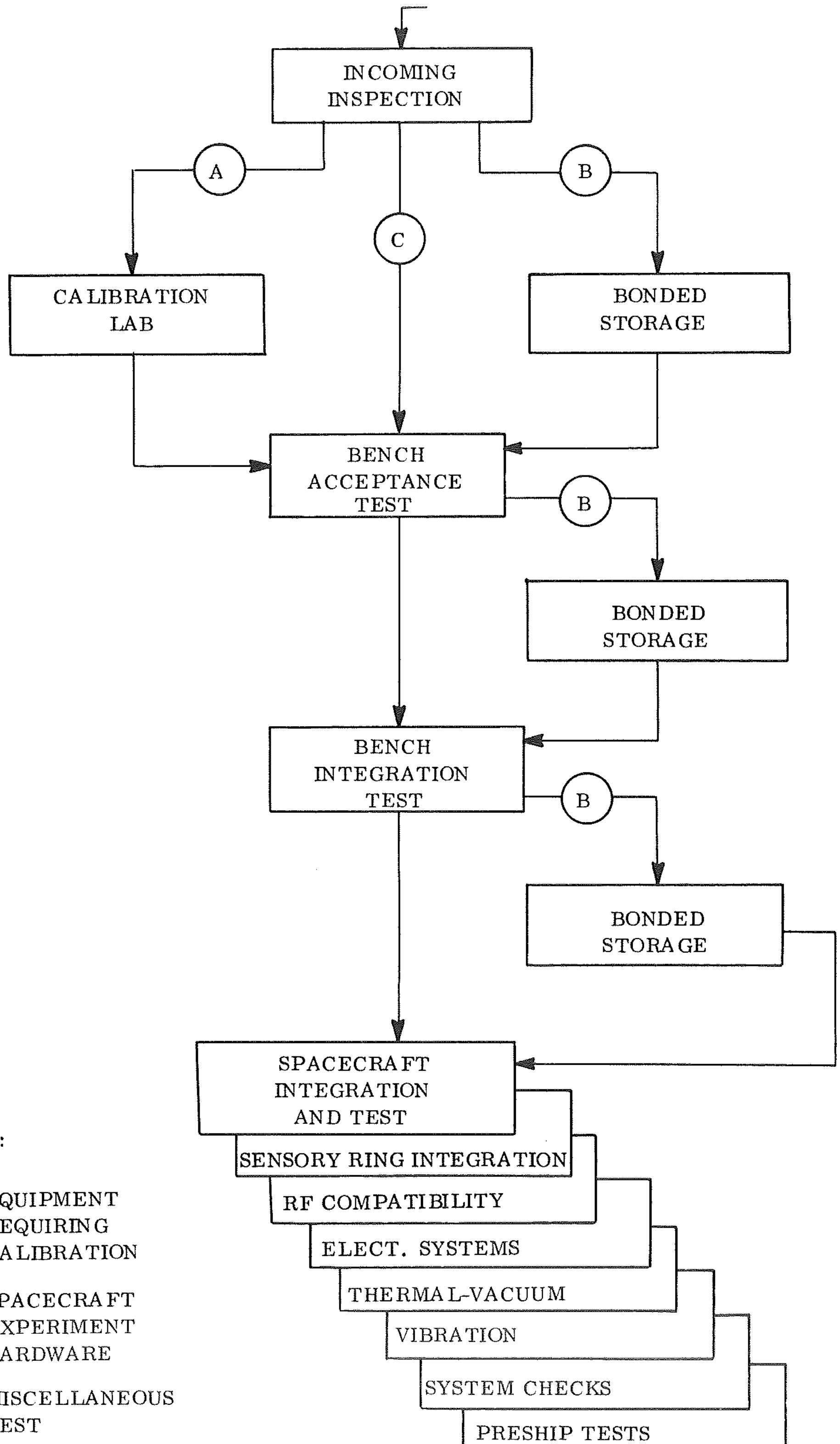
Shipping Containers

<u>SHIPPING CONTAINERS MUST BE MARKED "NIMBUS"</u>, and must indicate whether spacecraft hardware or support hardware or both are enclosed so that appropriate handling procedures are assured.

Importance of Identification

<u>THE IMPORTANCE OF EXPERIMENT HARDWARE SERIALIZATION AND MARKING</u>. and test support equipment marking, can be seen from the simplified equipment flow diagram at the spacecraft contractor's facility (opposite page). After incoming inspection, equipment will flow in and out of test areas. the calibration laboratory, and bonded storage many times. This movement of hardware from numerous experimenters and subsystem contractors requires strict serialization and marking to maintain hardware control.

EQUIPMENT FLOW



LEGEND:

A EQUIPMENT REQUIRING CALIBRATION

B SPACECRAFT EXPERIMENT HARDWARE

C MISCELLANEOUS TEST EQUIPMENT

SHIPMENT of HARDWARE and DOCUMENTATION

Advance Notice

Advance notice of all shipments must be provided to the spacecraft contractor to allow for advance preparation of receiving inspection documentation and facilities for accommodating the experiment hardware.

Advance notice must be made by TWX one (1) week prior to shipment and should note the contents of the shipment:

- NOMENCLATURE AND SERIAL NUMBERS of hardware being shipped
- LIST OF DRAWINGS AND DOCUMENTATION included in the shipment
- DESCRIPTION OF BENCH TEST EQUIPMENT included in the shipment
  - Date of shipment and mode of transportation
  - List of hardware
  - Approximate size and weight
  - Harnesses available with shipment
  - Manuals, schematics, etc.

The following documentation should accompany all hardware shipments:

- CALIBRATION CURVES
- CONNECTOR MATE AND DEMATE HISTORY for all prototype and flight hardware to be shipped.
- ACCEPTANCE LOG BOOK of assembly, inspection and test of all prototype and flight hardware to be shipped.
- OPERATING TIME SUMMARY listing total "Power On" time up to point of shipment.

Additional Provisions

- EACH PIECE OF EQUIPMENT AND DOCUMENTATION must be identified to ensure against loss or misuse.
- BENCH ACCEPTANCE TEST PROCEDURES are required no later than two (2) weeks prior to the initial shipment of experiment hardware to the spacecraft contractor.
- TEST PROCEDURES ARE REQUIRED EARLY to allow for spacecraft contractor test team review and familiarization.
- SPECIFIC NAMES AND ADDRESSES will be provided to experimenters by the spacecraft contractor for the purpose of addressing advance notifications of shipments.
- THE ENTIRE SPACECRAFT SCHEDULE IS DEPENDENT ON THE RECEIPT OF EXPERIMENT HARDWARE AT PLANNED POINTS THROUGHOUT THE PROGRAM. Failure to meet a shipment schedule or provide proper advance notice could jeopardize spacecraft delivery and launch.
- SHIPPING CONTAINERS must include shock absorbing provisions and packing features which protect the hardware from damage during all transit and handling conditions.
  - Dessicants must be included in containers
  - Contamination of hardware must be prevented.
- CONNECTOR PROTECTION must be provided on all hardware and test cables. Plastic caps or similar covers must be provided to ensure the safe handling of all electrical connectors.

SPACECRAFT CONTRACTOR FACILITY ACCESS

PERSONS CONCERNED WITH THE NIMBUS PROGRAM from GSFC or from contractor agencies are required to accomplish two steps in order to insure entry into the spacecraft contractor facility.

THE FIRST is to notify the spacecraft contractor's Nimbus Program Office of the desire to visit concerning Nimbus and to indicate the purpose of the visit. This, in terms of security, will establish the "need to know." Steps will be taken by the NASA/Nimbus Project Office upon receipt of this information to insure that the Security Officer, through the Nimbus Program Office, receives the request for visit indicating the time period involved. (Note that security information should be forwarded at least two weeks in advance.)

THE SECOND is to notify the Security Office at GSFC or the contractor's plant (in the case of co-contractors) of the intended visit and state the clearance required. In turn, this security request will normally be honored by the spacecraft contractor's Security Officer. The GSFC or co-contractor's Security Officer will notify the spacecraft contractor's Security Officer. This should be done in writing or by TWX.

RESIDENT OR EXTENDED VISIT - If an individual is planning to reside at the spacecraft contractor facility or to visit for an extended period, he should establish a security clearance for that period and can request a picture badge which will insure immediate entry. If no picture badge is obtained,a "no-escort" badge may be obtained at the visitor's desk. The "no-escort" badge requires the same clearance and "need-to-know" as mentioned above. Receipt of a picture badge requires the individual to be photographed by the spacecraft contractor's Security Office.

IN AN EMERGENCY, if an individual desires to visit concerning the Nimbus Program, entry into the facility can also be achieved by receiving a visitor's badge at the desk upon arrival and notifying the Nimbus Project Office at the spacecraft contractor facility. It is suggested that this practice be avoided because it involves having to be escorted at all times in the plant. This can be a hindrance to the timely completion of business during the trip.

WORKING NOTES

Page No.

Interface Meeting Notes .... 1 to 5
Experiment Information Checklist .... 6 and 7

INTERFACE MEETING NOTES

SUBJECT: ______________________________

MEETING DATE: ______________ PLACE: ______________

ATTENDANCE: ______________ ______________

______________ ______________

______________ ______________

______________ ______________

EXPERIMENT DESIGN STATUS:
PROBLEMS:
INTERFACE AGREEMENT ITEMS:
ACTION ITEMS:

INTERFACE MEETING NOTES

SUBJECT: ______________________________

MEETING DATE: ______________ PLACE: ______________

ATTENDANCE: ______________ ______________

______________ ______________

______________ ______________

______________ ______________

EXPERIMENT DESIGN STATUS:
PROBLEMS:
INTERFACE AGREEMENT ITEMS:
ACTION ITEMS:

INTERFACE MEETING NOTES

SUBJECT: ______________________________

MEETING DATE: ______________ PLACE: ______________

ATTENDANCE: ______________ ______________

______________ ______________

______________ ______________

______________ ______________

EXPERIMENT DESIGN STATUS:

PROBLEMS:

INTERFACE AGREEMENT ITEMS:

ACTION ITEMS:

INTERFACE MEETING NOTES

Subject: ______________________________

Meeting Date: ______________ Place: ______________

Attendance: ______________ ______________

______________ ______________

______________ ______________

______________ ______________

Experiment Design Status:
Problems:
Interface Agreement Items:
Action Items:

INTERFACE MEETING NOTES

SUBJECT: ______________________________

MEETING DATE: ______________ PLACE: ______________

ATTENDANCE: ______________ ______________

______________ ______________

______________ ______________

______________ ______________

EXPERIMENT DESIGN STATUS:

PROBLEMS:

INTERFACE AGREEMENT ITEMS:

ACTION ITEMS:

EXPERIMENT INFORMATION CHECKLIST

EXPERIMENT: ______________________________

EXPERIMENTER: ____________________ PHONE NO. ____________________

NASA EXP. T.O. ____________________ ____________________

CONTRACTOR REPS. ____________________ ____________________

____________________ ____________________

SENSING TECHNIQUE:							
CHANNELS/FREQ.							
MEASUREMENT OBJECTIVES							
RESOLUTION/ACCURACY							
DUTY CYCLE	ON:		OFF:				
FIELD OF VIEW/BEAMWIDTH							
SCANNING							
TOTAL POWER	PEAK		STANDBY		AVE. ORBITAL		
COMPONENTS	SIZE		WEIGHT		AVE. POWER		
		E		E		E	
		C		C		C	
		A		A		A	
		E		E		E	
		C		C		C	
		A		A		A	
		E		E		E	
		C		C		C	
		A		A		A	
		E		E		E	
		C		C		C	
		A		A		A	
		E		E		E	
TOTALS		C		C		C	
		A		A		A	

E - ESTIMATED C - CALCULATED A - ACTUAL

EXPERIMENT INFORMATION CHECKLIST

Structural/Mechanical

Special Temp. Rqmts.

Pressurized Units
Special Install Rqmts

Special Location Rqmts

Alignment Rqmts

Special Accessibility

Ambient Targets

T/V Targets

Command/Clock

No. of Commands
Special Sequences
Clock Frequencies

Stored Commands

Telemetry

Analog Monitors
Digital A Monitors
Digital B Monitors
Special Calibration Modes

1/1 Sampling Rates
1/16 Sampling Rates
Other

Sensor Output Data

Channels	Analog or Digital	Bit Rates	Freq. Resp.	Voltage Levels	Source Impedances